

AUTONOMOUS SPACE PROCESSOR FOR ORBITAL DEBRIS
Final Report
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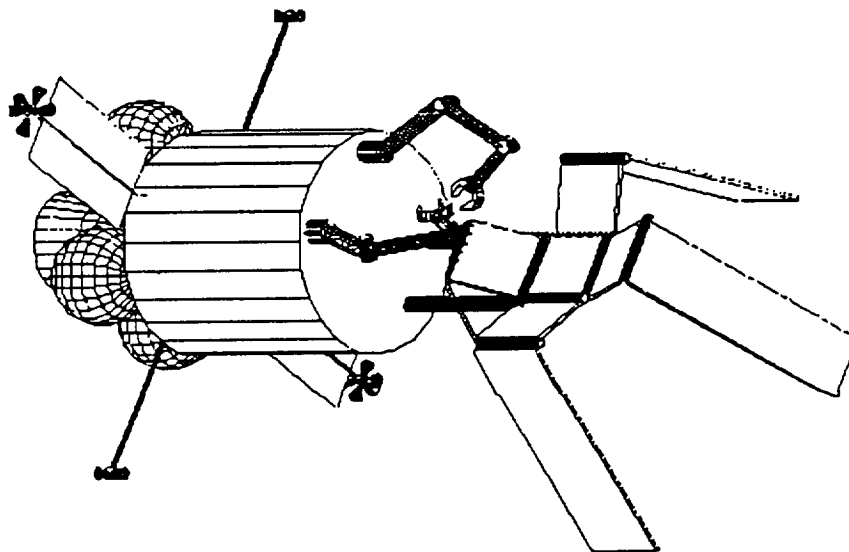


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ADVANCED DESIGN FOR ORBITAL DEBRIS REMOVAL IN SUPPORT OF SOLAR SYSTEM EXPLORATION

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Abstract

This work continues to develop advanced designs toward the ultimate goal of a GETAWAY SPECIAL to demonstrate economical removal of orbital debris utilizing local resources in orbit. The fundamental technical feasibility was demonstrated last year through theoretical calculations, quantitative computer animation, a solar focal point cutter, a robotic arm design and a subscale model. During this reporting period, several improvements are made in the solar cutter, such as auto track capabilities, better quality reflectors and a more versatile framework. The major advance has been in the design, fabrication and working demonstration of a ROBOTIC ARM that has several degrees of freedom. The functions were specifically tailored for the orbital debris handling. These advances are discussed here. Also a small fraction of the resources were allocated towards research in flame augmentation in SCRAMJETS for the NASP. Here, the fundamental advance has been the attainment of Mach numbers up to 0.6 in the flame zone and a vastly improved injection system; our current work is expected to achieve supersonic combustion in the laboratory and an advanced monitoring system. David Andrus and Gordon Ingmire are the students working on SCRAMJETS.

Introduction

We can hardly improve upon the lucid descriptions of the Orbital Debris issue, by science writers¹⁻⁴ and other popular news media coverage⁵⁻⁹. Without doubt the problems of orbital debris have grown to be of serious concern to astronomers, space technologists and to terrestrial dwellers. The specific problems were presented at the XXXIX IAF Congress. The University of Arizona Space Engineering Design team is developing the design for economical removal of the larger debris pieces through local resource utilization. The fundamental idea is to concentrate solar energy into a point focus, cut the debris into precise shapes that can be added on to the "sweeper" craft and robotically assemble the pieces into a manageable configuration--to be followed by one of three disposal modes: (i) retrieval by a spacecraft (STS, HERMES, BURAN, ...), (ii) precise splashdown into the oceans, or (iii) planned burnup during atmospheric reentry. The fundamental space technologies to be demonstrated are solar cutting of candidate space junk materials, robotic assembly and accurate disposal. In 1988 the University of Arizona began participation in the USRA program and demonstrated solar cutting and a subscale model robotic arm. This year, 1989, a full scale robotic arm has been constructed, is operational and the entire assembly is shown to be technically

feasible. This report is a summary of the work and explains the details of the space engineering.

Consistent with the USRA philosophy, new undergraduate students were brought on board. This year, five new students were involved in the Autonomous Space Processor for Orbital Debris (ASPOD) design, two new students built a "target" spinning satellite and two other students worked on advanced designs for the NASP. The project continues to draw worldwide attention, including letters from elementary and high schools. Some of these are documented in the Appendix.

The support from USRA and technical monitoring of Mr. James D. Burke of JPL are gratefully acknowledged. Mr. Milton Schick contributed towards the new robotic arm.

Robotic Arm Design

Design Specifications

The ASPOD design incorporates a solar powered metal cutter to facilitate dead satellite processing in a cost effective manner. In order to position debris at the focal point it is necessary that the ASPOD be equipped with robotic arms. The final spacecraft will require two arms to insure any final movement imparted to the debris will not cause the cut piece to move toward the key lenses and mirrors of the solar cutter. The task for this year's team, however, is the design and fabrication of one arm to be used in conjunction the current solar cutter. The arm is to hold and move material to be cut in the focal plane of the solar concentrator. The design will have to meet several requirements. The movement of the arm should not interfere with the light being concentrated by the focal point cutter. Hydraulic actuators cannot be used due to the fact that hydraulic fluid leakage would be difficult to control in the hard vacuum of space. The arm should be able to position curved and flat pieces of material so that the concentrated light is perpendicular to the surface being cut. To insure accurate control of the arm, tight tolerances (0.002 inch on all critical load bearing members) are observed. This is evident in the lack of play in the arm.

All modes of operations of the arm, including deploy/stow, processing and placement require slow steady motion. The current solar concentrator will cut 0.005 inch aluminum at approximately 5 inches/minute. This is the nominal speed at which the arm is designed to operate.

Final Design

A sketch of the arm is presented in Figure 1 showing major components and specifications. Each component is described in detail below. The design work was carried out in six coordinated sections: the end effector, the grip actuator, the wrist (both bending and rotation), elbow and forearm segment, upper arm rotation and shoulder swivel.

End effector

The end effector is of a simple claw type design. Two 5 1/8 inch radius arc shaped fingers allow grasping of large objects of varying geometries. At the end of the fingers are knurled surface grips that swivel to allow objects with non parallel faces to be grasped. The fingers and grips are made of 1/16 inch 6061-T6 aluminum and have hollow sections that give the strength needed as well as minimizing the weight. Figure 2 shows finger and tip geometry. The fingers were designed based on the clamping force needed to hold a mock satellite. This mock satellite was sized in proportion to the current solar cutters heat flux versus that which is estimated to be required for the full size solar cutter; approximately 1/10 scale.

Grip actuator

The fingers are designed to operate at a rotation rate of 0.105 rad/sec. Based on the force needed to hold the mock satellite, and the geometry of the fingers the maximum torque required was determined to be 1.3 ft-lbs. The drive motor for the fingers, as well as the other arm joints, is a 24 volt DC

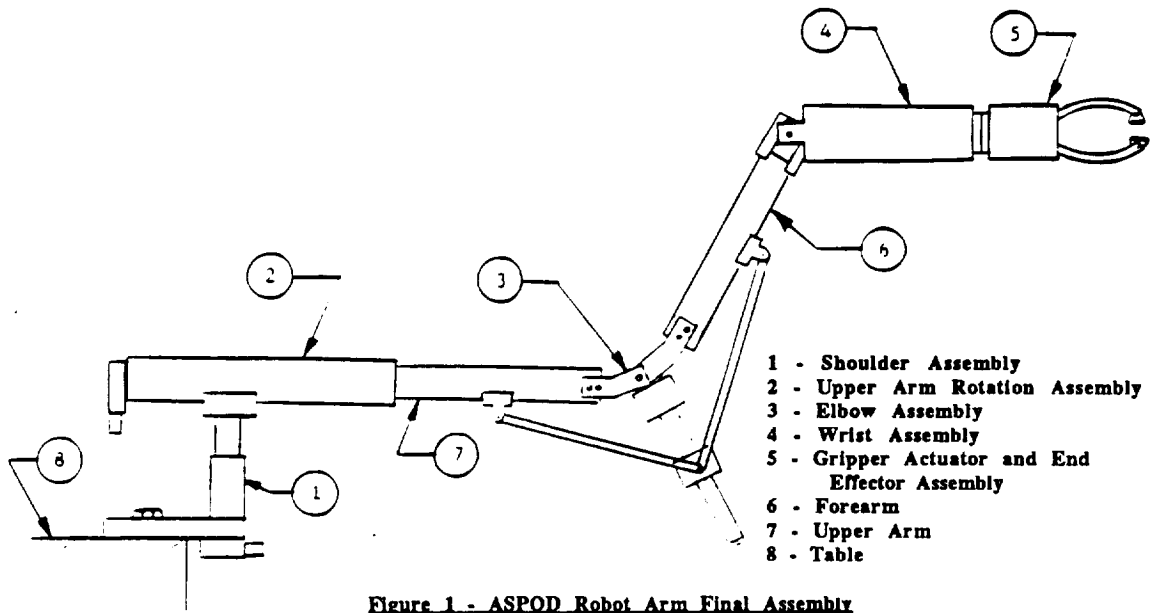


Figure 1 - ASPOD Robot Arm Final Assembly

SPECIFICATIONS:

Degrees of Freedom	5
Approx. Weight	25 pounds
Max. Horizontal Reach	66 inches
Max. Vertical Reach	59 inches

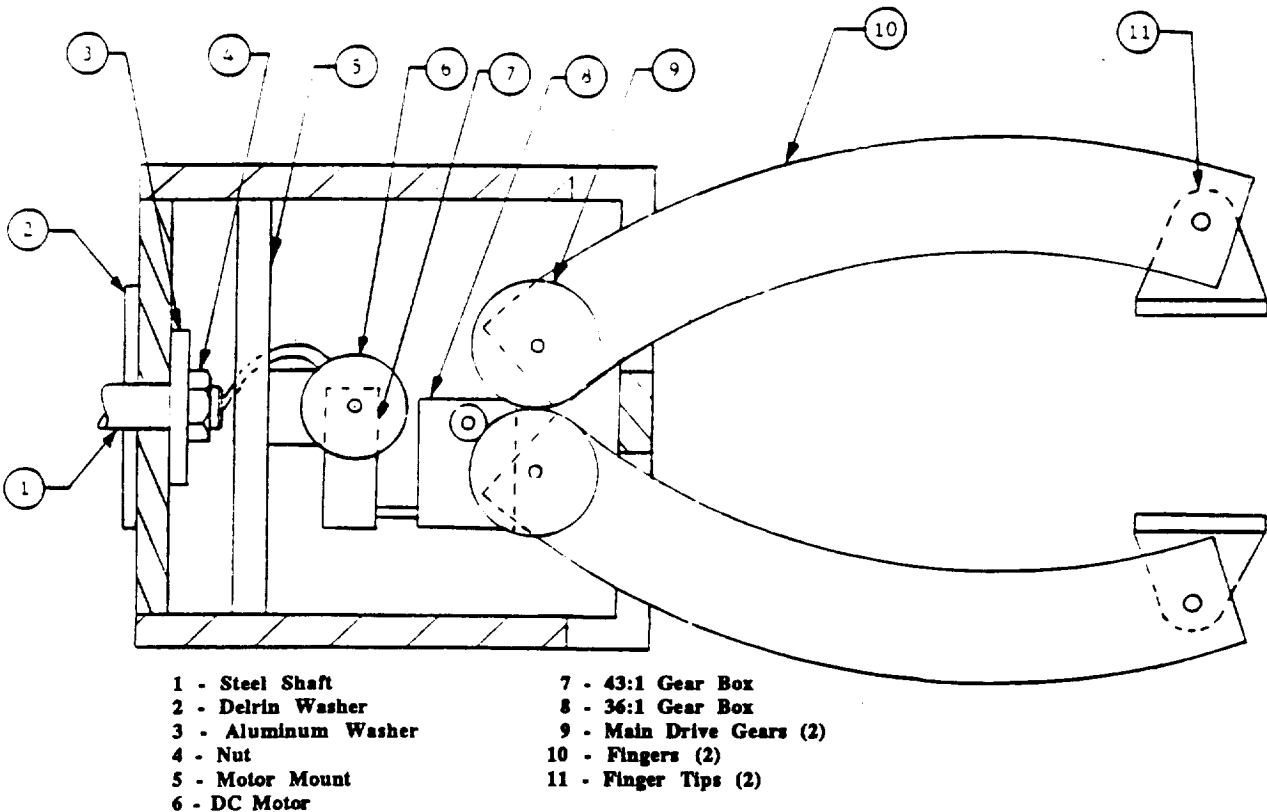


Figure 2 - Gripper Actuator and End Effector Assy.

motor coupled in series to a 43:1 and 36:1 reduction gearboxes. See figure 2 for the grip actuator assembly.

Wrist assembly

The wrist has two degrees of freedom: rotation and bending. The wrist can rotate a full 360 and bend 90 degrees in both directions. The wrist housing is made of linen phenolic composite and the inner component supports are composed of aluminum and polycarbonate. The wrist rotation assembly is shown in Figure 3. It is powered by a 24 volt dc motor coupled to two 36:1 reduction gearboxes with a 5:1 final pinion and spur, producing 474 in-oz at one rpm. The required torque was determined by assuming the wrist must counter a moment produced by the mock satellite if grappled at one end in a cantilever fashion. The bearings between the gripper and wrist housings are made of delrin, having a coefficient of friction comparable to teflon but with greater rigidity. Furthermore delrin was selected over ball bearings because of the appreciable weight savings. The bearing shaft was attached to the grip actuator housing and was bored out to accommodate the wires for the grip actuator motor. To eliminate twisting of the wires when the wrist rotates a slip ring mechanism is attached to the final spur gear.

Wrist bending is powered by a 24 volt dc motor with a 550:1 reduction gear train contained in the wrist housing. The final drive shaft is attached to the forearm and is extended through both sides of the wrist housing.

Arm and elbow

The forearm and upper arm segments are made of 6061-T6 aluminum and were sized according to the mock satellite load described above. The elbow mechanism incorporates an aluminum square thread power screw as shown in figure 4. The power screw points outside the elbow, keeping the length to eight inches while still allowing 175 degrees of forearm rotation about the upper arm. Movement is provided by a 24 volt DC motor running through a 337:1 reduction gearbox. The collar power nut is machined from delrin, and is clamped between an aluminum collar operating pushrods connected to the upper and forearm sections.

Upper arm rotation and shoulder swivel

A sketch of the upper arm rotation assembly is shown in figure 5 and the shoulder swivel assembly is shown in figure 6. Similar designs are used for both which allow a full 360 degrees of rotation. The inner tube extension is made of 1 1/4 inch schedule 40 6061-T6 aluminum tubing. The loadings that this section must handle include torsional, bending, and compression and tension. Nylon is used as an inside bearing surface, and delrin filled aluminum thrust washers were manufactured to reduce friction and power requirements.

The bearing housing consists of 2 1/4 inch schedule 40 6061-T6 aluminum tubing. Delrin is used for the outer bearing surface, which mates with the above mentioned nylon bearings producing a low friction, light weight bearing assembly capable of handling large loads.

The gearbox assembly produces a 17500:1 reduction and is driven by a 24 volt DC motor.

Photographs of the assembled arm are shown in Figure 7.

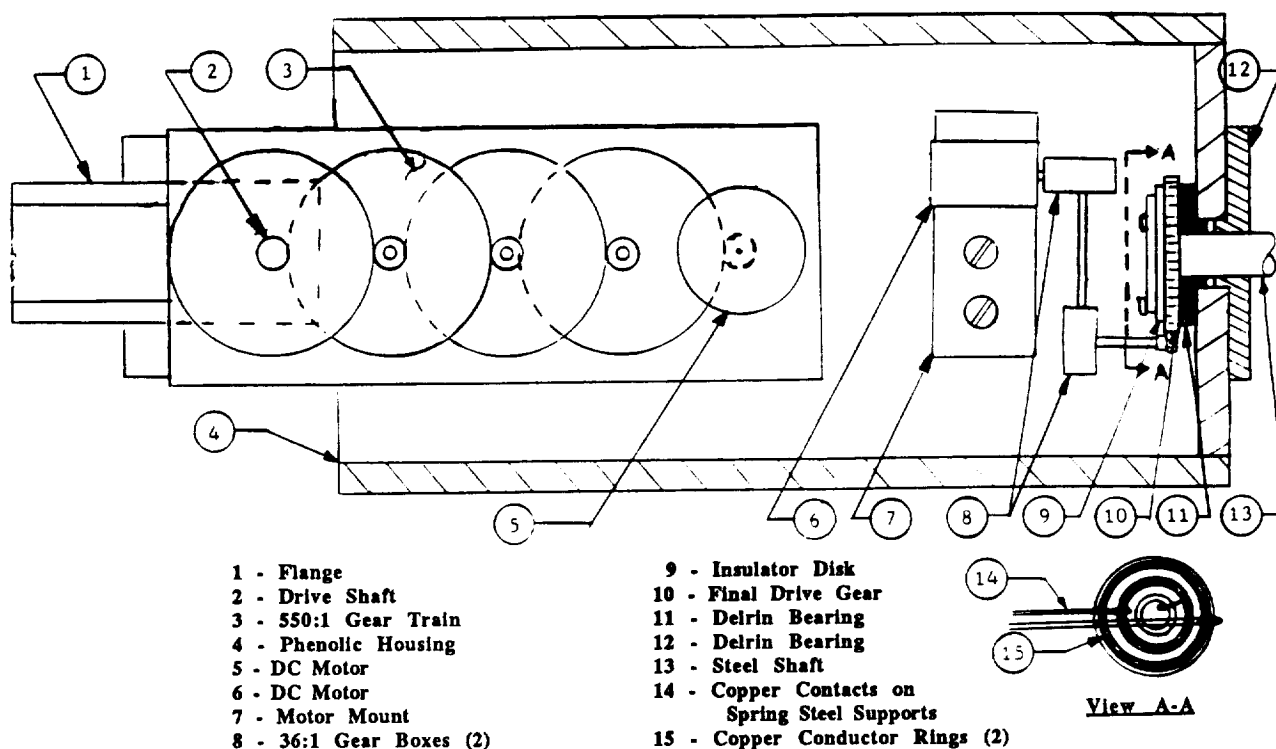


Figure 3 - Wrist Assembly

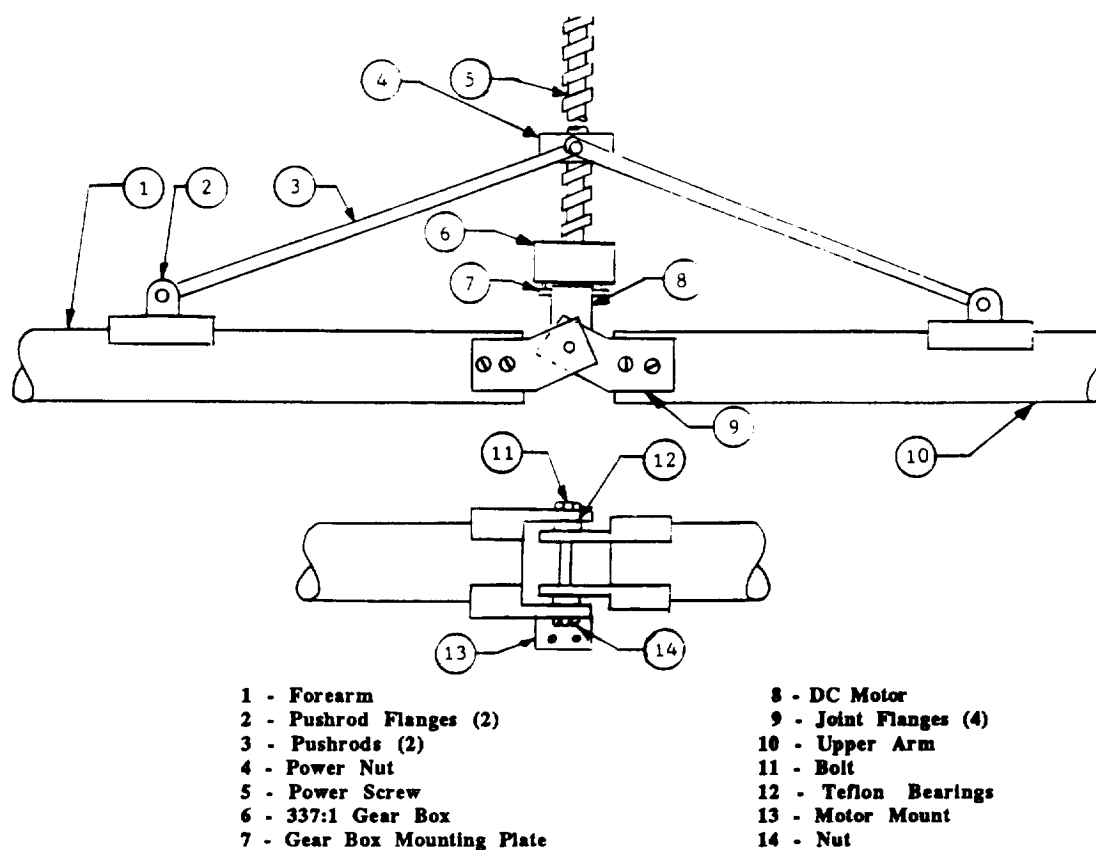


Figure 4 - Elbow Assembly

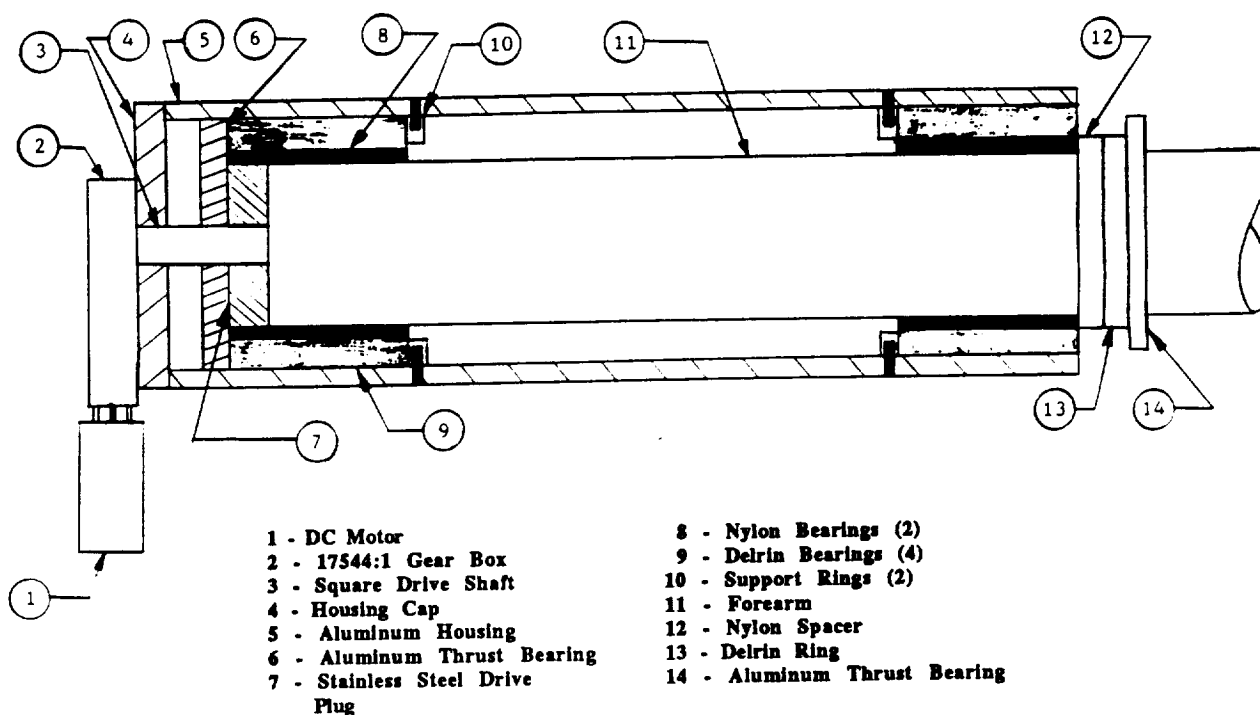


Figure 5 - Upper Arm Rotation Assembly

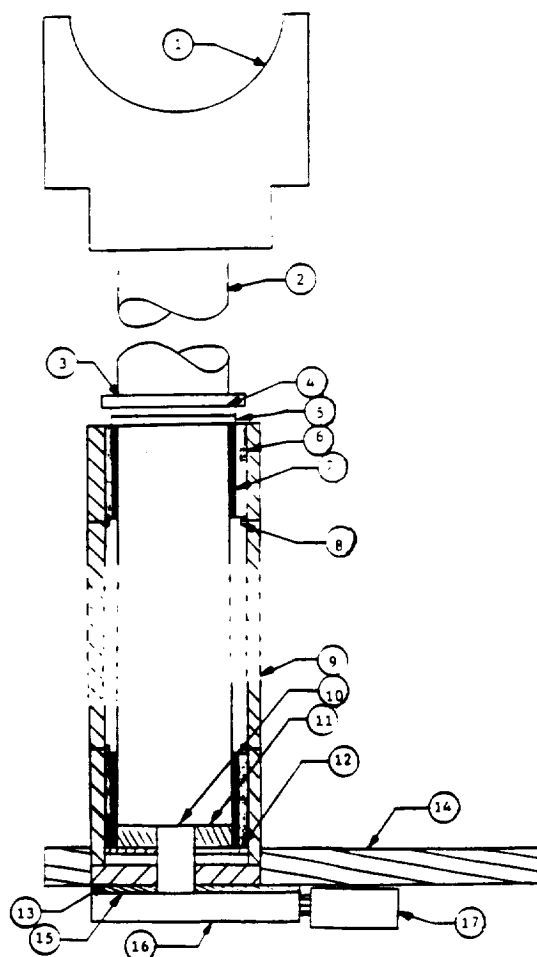


Figure 6 - Shoulder Assembly

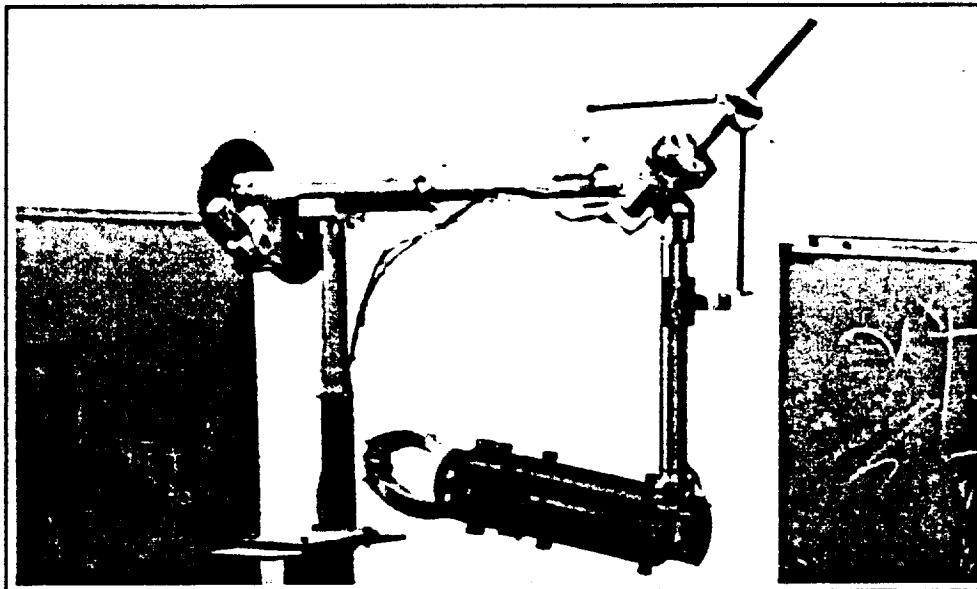


Figure 7a Assembled arm



Figure 7b. Assembled system cutting
a piece of metal at the focal point

Solar Tracker

The solar tracker consists of a custom made mount and circuitry controlling two degrees of freedom. It provides directional positioning of the table mounted focal-point-cutter during operation. There are two independent control/drive circuits, one for elevation, the other for azimuth. Figure 8 shows a block diagram of the major components.

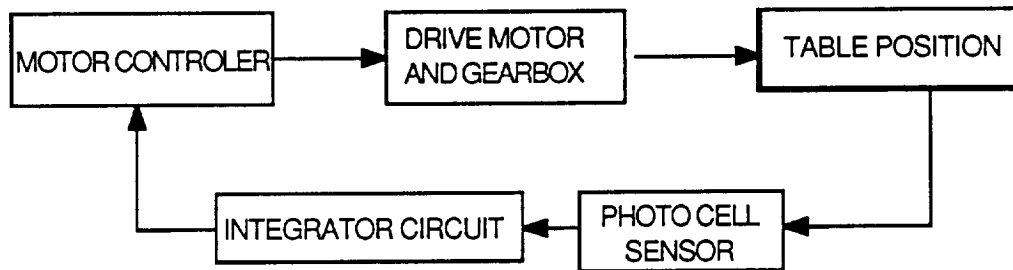


Figure 8. Block diagram of tracker components.

The position of the sun is determined by a sensor composed of two photo cells mounted at 90° . This sensor is mounted facing the sun so that the bisector of the angle between the photo cells coincides with the optical axis of the focal-point-cutter. The sensor and integrator circuit are wired as shown in Figure 9 so that when the cutter is pointing directly at the sun the sensor output is zero and the integrator circuit output (V_{out}) is constant. This constant voltage is input to a motor speed controller. When the cutter begins to drift off track the sensor output becomes nonzero signaling the controller to decrease or increase motor speed accordingly.

The solar tracker has been tested and maintains positioning accuracy to within approximately $\pm 1^\circ$.

Photographs of the components and assembled system are shown in Figure 10

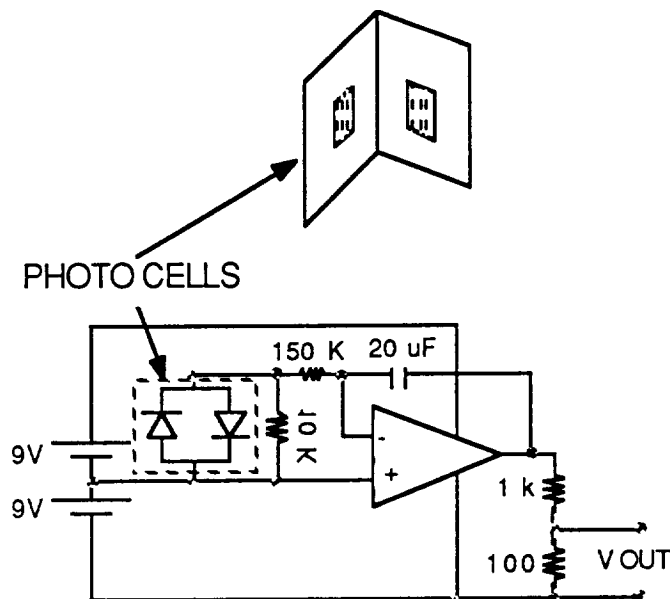


Figure 9. mounted photo cells and arrangement in integrator circuit

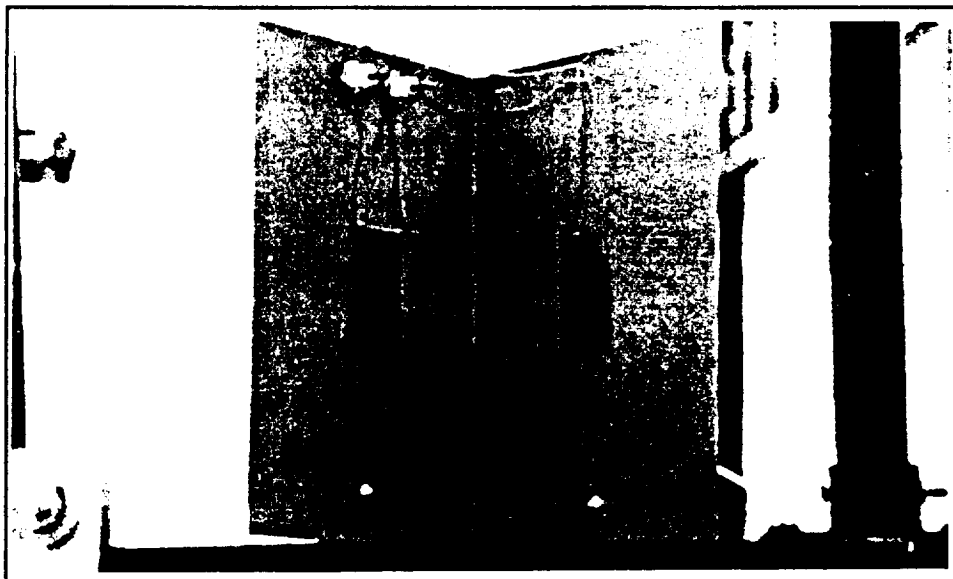


Figure 10 a: Photo cell sensor.

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Figure 10 b: Measurement of temperature at focal point.

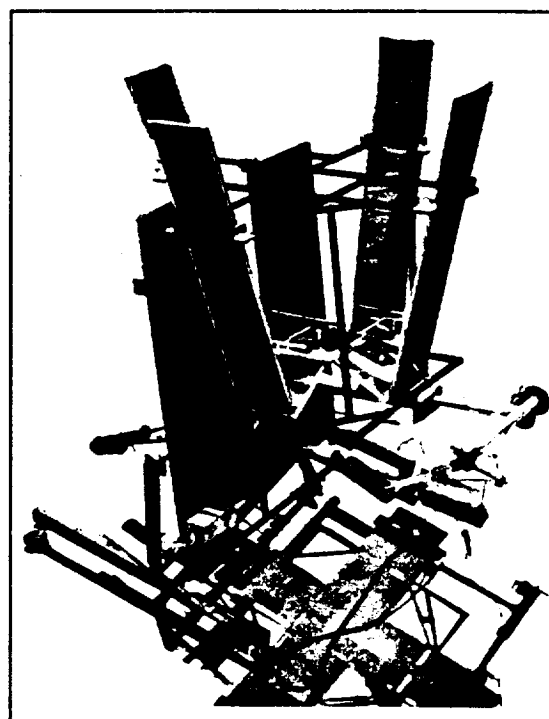


Figure 10 c: System operating with auto tracker.

Mission Scenario Selection

ASPOD may be delivered to the rendezvous orbit via STS or an expendable launch vehicle. A schematic of the mission sequence is presented in Figure 11. Once ASPOD has rendezvoused with a target, the question becomes: what to do with it? Several different debris disposal scenarios were looked at. They included:

1. Retrieve and carry along each debris, rendezvous with shuttle for salvaging.
2. Aerobraking; deploy large mylar sheets on debris to increase rarified atmospheric drag, thus lowering orbital lifetime.
3. Attach a small solid rocket motor (SRM) to transfer debris to a lower lifetime orbit.

Two methods of orbital transferring were investigated:

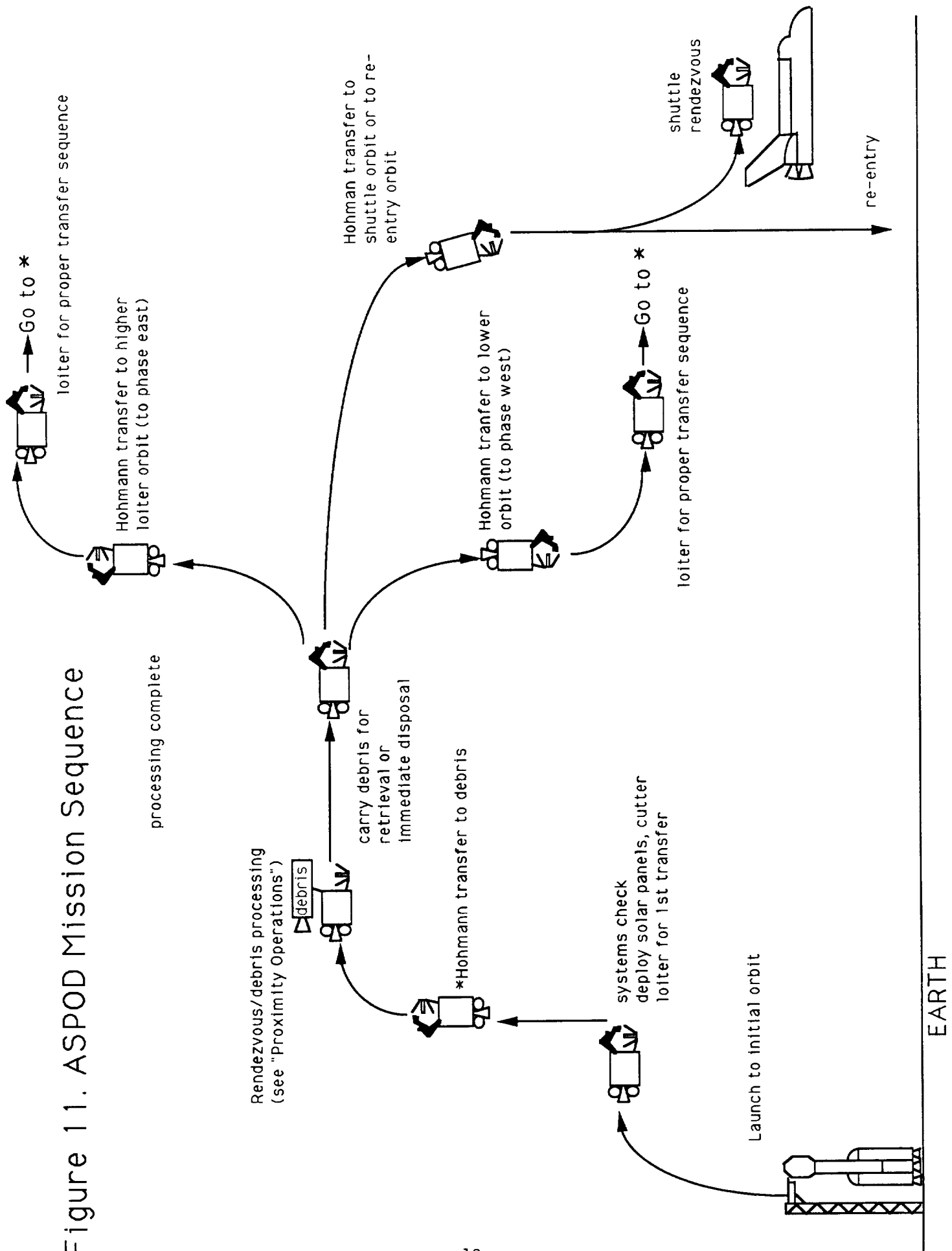
1. ΔV for direct transfer between orbital planes.
2. Using the natural orbital perturbations from the oblateness of the Earth, for nodal regression to synchronize orbital planes.

These are discussed further in the Sample Mission section.

Some satellites now in Earth orbit may be worth bringing down for scientific studies or salvaging. Some debris might hold a wealth of scientific information concerning long term satellite exposure to micrometeorites and small debris. Some satellites may contain expensive materials, parts, or top secret data or machinery. Thus, for this particular type of debris, ASPOD would cut them with the solar cutter and store them in an onboard storage bin. With each retrieval, ASPOD will increase in mass, compared to other spacecraft which tend to decrease in mass. This will raise the propellant requirement considerably. However, with careful planning, a retrieval mission could still be quite successful and is considered in detail in the Sample Mission section.

Aerobraking uses the rarified atmosphere to cause drag on a satellite to lower its orbit or change its orbit in a hyperbolic planet fly-by. Orbital lifetime is a function of altitude, eccentricity, mass, and area perpendicular to the direction of travel. Thus, if one were able to increase this area, it would be possible to decrease the orbital lifetime of a piece of debris, allowing it to disintegrate and burn up in the atmosphere upon reentry. To analyze this, a 1 year lifetime was assumed and the necessary area was calculated for a typical 2000 kg piece of debris in a circular orbit of 750 km. This was found to be $2,367\text{m}^2$, which corresponds to a sheet of mylar, 48m X 48m with composite structural supports behind to hold it rigid. In the lower atmosphere, the drag on a sheet this large will become very high requiring a stronger and heavier structure. Due to the expected high mass, difficulties in attitude control, increased target area for smaller debris, and unreliable estimates of the upper atmosphere for aerobraking this method was not analyzed further.

Figure 11. ASPOD Mission Sequence



Small solid rocket motors are considered as a means of de-orbiting debris after rendezvous and processing. The SRM's would be welded to payloads by the solar cutter and would be attached to boosters by inserting them into the nozzles of the rocket motors. ASPOD will then use its attitude control thrusters to orient and spin-up the debris for the SRM burn. This burn will result in a one ΔV orbit transfer (see figure 12).

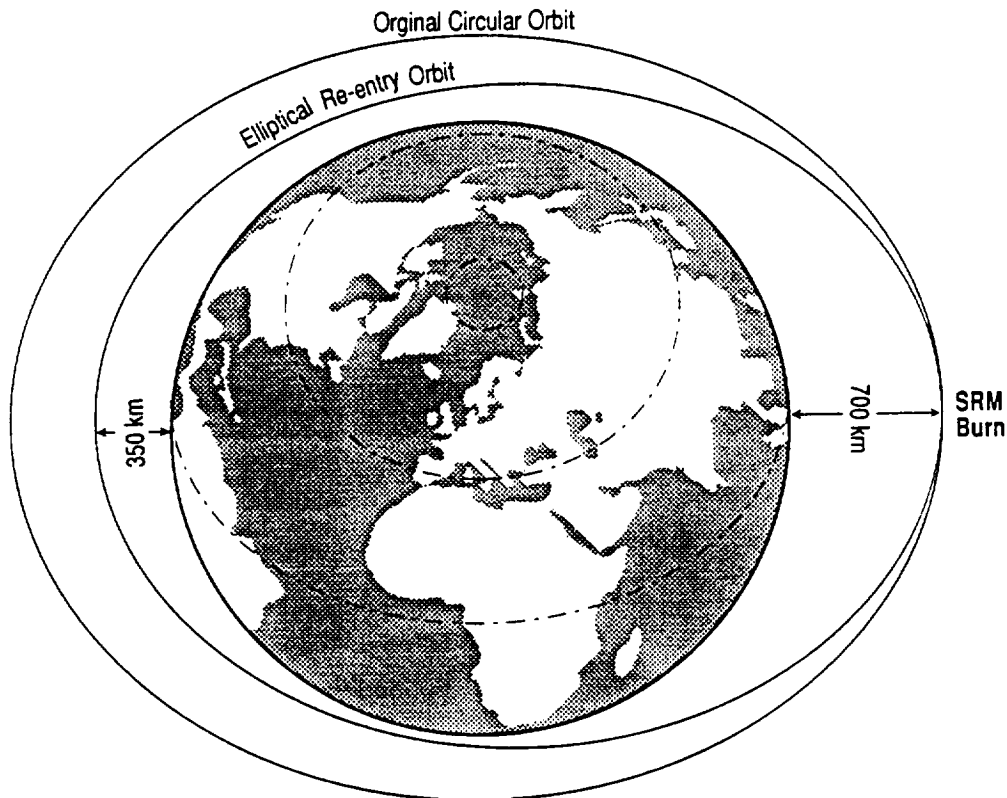


Figure 12. Debris Re-entry Trajectory

The point of this burn will become the apogee of a new elliptical orbit. The perigee altitude will be 350 km to allow for a reasonably rapid orbital decay.

For each piece of debris to be de-orbited, an SRM has to be brought along. In addition, the propellant needed to transfer to each debris must also be brought along. Thus, as the number of pieces of debris targeted increases the launch weight of the satellite will increase exponentially. A numerical example is discussed in the Sample Mission section.

After all the chosen debris has been de-orbited, ASPOD will fire its engine consuming all remaining propellant for a reentry orbit. This prevents ASPOD itself from contributing to the debris problem.

Baseline Target Selection

In choosing which debris to remove from orbit, there are many considerations. The most important are probability of collision, mass, orbital lifetime, and accessibility. Thus, the search concentrated on locating large

pieces of debris with long orbital lifetimes. Since the majority of debris resides in altitudes between 700 and 1600 km¹⁰, objects not in this range were ruled out. The search was also limited to objects with an angle of inclination below 50° to minimize propellant requirements. It would be possible to achieve a rendezvous with a target above this range, but with an abundant amount of debris below, such targets were not currently baselined.

Larger, more massive pieces of debris are bigger targets and have more kinetic energy. When they are struck, they scatter a greater amount of small debris. This small debris is what concerns NASA and NORAD the most. From studies done on hypervelocity impacts, a small piece of debris, on the order of 3% mass relative to the target, can turn the target into shrapnel¹¹. Thus an exploding rocket booster or a satellite break up from an impact could add to the debris in earth orbit by the thousands. These small pieces are also very difficult to track. NORAD is currently capable of tracking debris larger than 10 cm at 1,000 km¹². Thus, it would be appropriate to remove a larger object before it becomes many small objects.

A simple yet important aspect in choosing the target debris was orbital lifetime. Some objects, because of their low altitudes, will soon reenter and burn-up in the earth's atmosphere. It was assumed that ASPOD will not be launched before the turn of the century. This led to the decision to not go after any object that will reenter before the year 2025.

Because of the extreme altitudes and low probability of collision of geo-synchronous satellites, these were not included in the search for possible debris. These satellites actually present the lowest danger since they are significantly beyond most other satellites. At this time the debris population density at the geo-synchronous altitude is still relatively low. Also, since objects in geo-synchronous orbit are all at similar altitudes and inclinations ($i \sim 0^\circ$) their relative velocities approach zero.

Using the above criteria, over 200 pieces of debris qualified as potential targets. Out of these, four were eventually chosen to perform mission feasibility studies. The four pieces are all U.S. contributions to the debris population and are orbiting at altitudes of approximately 750 km and inclinations of 35°. These are used for a representative sample mission in the following section.

Orbital Techniques and Sample Mission

There are several inclinations in which a majority of the larger pieces of debris (roughly 2000 kg) are in orbit. The Soviet Union has put a large number of satellites into orbits with inclinations of 65°, 74°, and 82°, and altitudes ranging between 600 km and 1500 km. Most U.S. debris is located in orbits with inclinations of 21°, 28° and 35°. The pieces of debris in the 21° and 28° inclinations often have apogees greater than 36,000 km, while the pieces in the 35° inclination have orbits with altitudes of approximately 750 km. Because a majority of the large pieces of debris are located in only a few inclinations, the opportunity exists to "clean up" space an inclination at a time. Over the past three decades, many thousands of satellites have been launched into Earth orbit, with most of these having unique ascending node locations. Thus, while many satellites may have common inclinations, they are not necessarily in a common plane. Therefore, a method of transferring longitudinal planes, while minimizing the total ΔV and the propellant requirement, must be found.

Two methods of transferring from one longitudinal location to another, while remaining in the same inclination, were investigated. The first method that was investigated utilized spherical trigonometry to calculate the angle between orbital planes with common inclinations and different ascending node locations. Knowing the angle between the planes and the velocities of the satellites at the line of intersection, the ΔV 's were calculated. As was expected, this transfer profile was very expensive in terms of propellant requirements. Figure 13 shows the propellant mass versus total longitudinal difference ($\Delta\Omega$) curves for rendezvous and retrieval of 2-5 pieces of 2000 kg debris with inclinations of 35° and altitudes of 750 km.

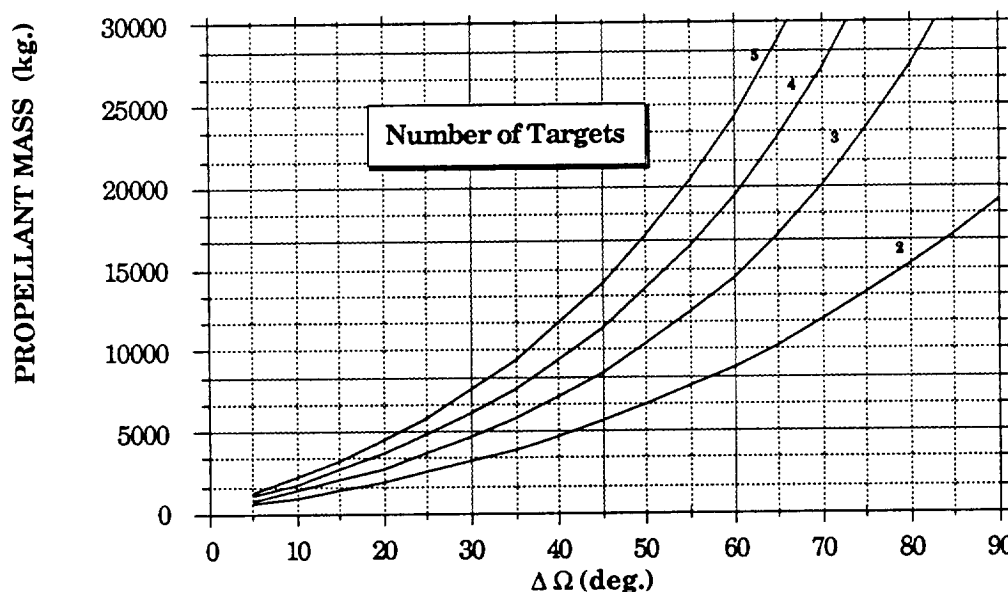


Figure 13. Propellant Mass vs. $\Delta\Omega$ and Number of Targets

As can be seen, the propellant mass requirements become too great for even modest ascending node differences (maximum Shuttle payload is approx. 16,000 kg). For all mass calculations an estimated dry mass of 526 kg (minus fuel tanks) was used with a main thruster specific impulse of 325 seconds. Because the size of the fuel tank is a function of the amount of propellant used, the mass of the fuel tanks was not included in the dry mass estimate. To take into account the effect of the tank mass on the propulsion requirements, a tank to fuel mass percentage was used. With a main thruster chamber pressure of 145 psia, a percentage of 2.3% was found to give a factor of safety for aluminum fuel tanks of 1.5.

If the Earth were a perfect sphere, an ASPOD mission to remove a significant amount of mass from low Earth orbit (LEO) would be seriously limited to pieces of debris that have longitudes of ascending nodes extremely close together. Because of the oblate shape of the Earth and the non-uniform gravitational field, natural perturbations exist, which, over time, significantly effect the orbital parameters of a satellite in LEO. The oblateness of the Earth particularly affects the ascending node of an orbit by applying perturbing torques on a satellite as it travels around the Earth¹³. This natural perturbation causes the locations of the ascending node to regress westward if the satellite

has a direct orbit ($0^\circ < i < 90^\circ$), and eastward if the satellite has a retrograde orbit ($90^\circ < i < 180^\circ$). Figure 14 shows the nodal regressions per day as a function of altitude and inclination (for circular orbits).

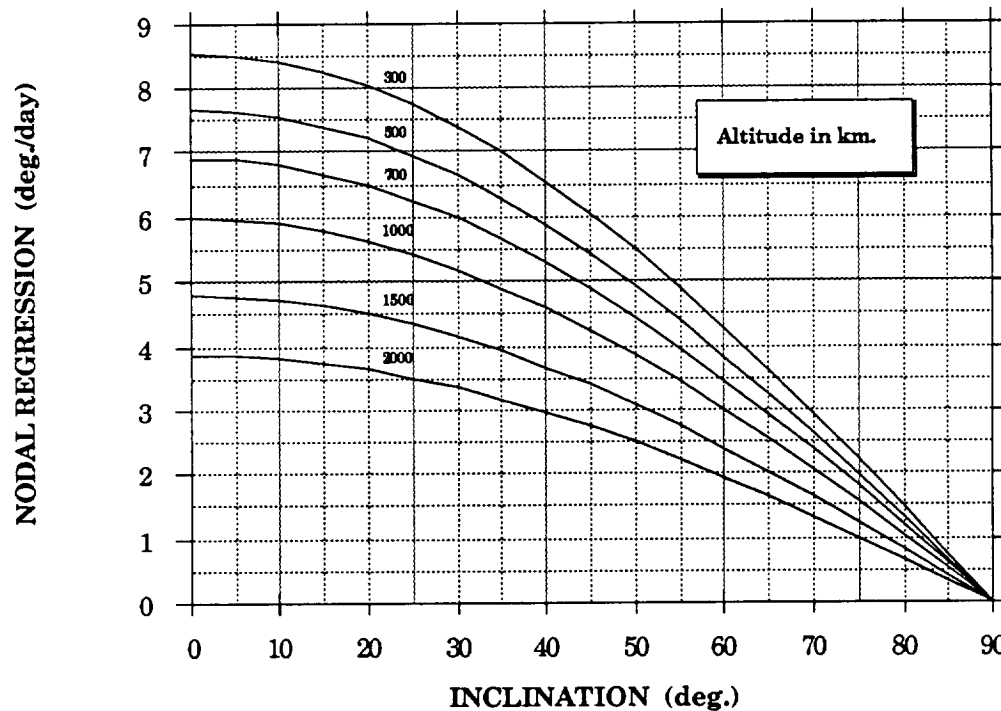


Figure 14. Nodal Regression vs. Inclination and Altitude

The second method of orbital transferring that was analyzed takes advantage of the natural nodal perturbations. Since the nodal regression rate at a given inclination is primarily a function of altitude, by establishing an ASPOD orbit at a different altitude than the debris orbit, the planes of the two orbits will eventually coincide. Therefore, the need for a plane change maneuver is eliminated and a coplanar Hohmann transfer can be employed.

If a piece of debris is to the west of ASPOD, establishing an orbit at a lower altitude will allow ASPOD to eventually align itself in the same plane as the debris. Likewise, if a piece of debris is to the east of ASPOD, loitering in a higher altitude orbit will eventually align the orbit planes. If ASPOD's ascending node differs by 180° than that of a debris orbit, ASPOD has the choice of either decreasing or increasing the altitude of its orbit to initiate the transfer sequence. For purposes of this analysis the criteria was established to loiter at an altitude that will result in a difference in nodal regression rates between the ASPOD orbit and the target orbit of at least 0.5 deg./day. Thus, ASPOD will not have to loiter for more than one year before an alignment of orbital planes will occur. Figure 15 demonstrates the propellant mass requirements using nodal regression for the retrieval of 2-6 pieces of 2000 kg debris with inclinations of 35° and altitudes of 750 km.

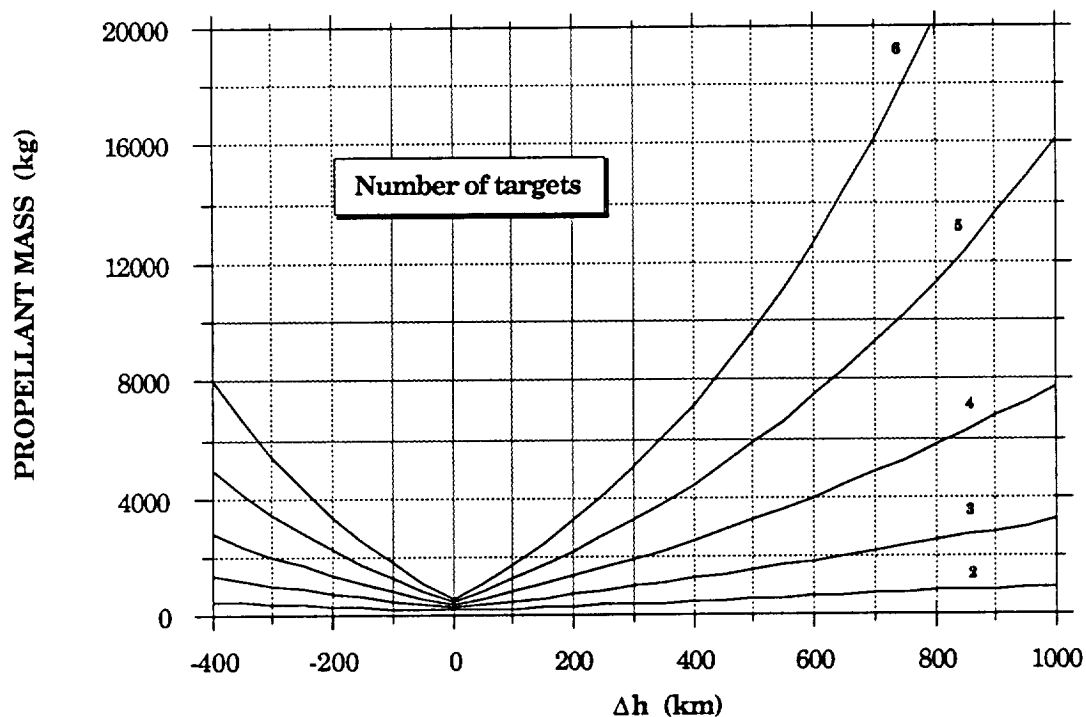


Figure 15. Propellant Mass vs. Δh and number of targets for two to six piece retrieval mission (2000 kg each). Note Δh determines loiter time.

Adhering to the criteria of a 0.5 deg./day minimum for nodal regression difference results in loiter orbit altitudes of 550 km and 950 km for a debris orbit of 750 km, or Δh 's of -200 km and +200 km, respectively. Since the ΔV 's are not dependant on the longitudinal differences, the propellant masses were plotted against Δh . From figure 15, using an ASPOD dry mass of 580 kg and specific impulse of 325 seconds, the required propellant mass to rendezvous with and retrieve four typical 2000 kg pieces of debris is approximately 1500 kg. For de-orbiting with SRM's rather than retrieval figure 16 shows the corresponding propellant mass for 2-6 pieces of large debris. From this figure, the necessary propellant to deorbit the chosen debris is approximately 200 kg. The four SRM's would add a total of 280 kg (70 kg each) to the initial dry mass of the ASPOD spacecraft.

The use of nodal regression to transfer orbits will now be applied to the retrieval of four actual pieces of debris. The orbital parameters and masses for the chosen debris as of Feb. 8, 1989 are shown in table 1 below (longitude location of OAO-2 telescope is used as reference location).

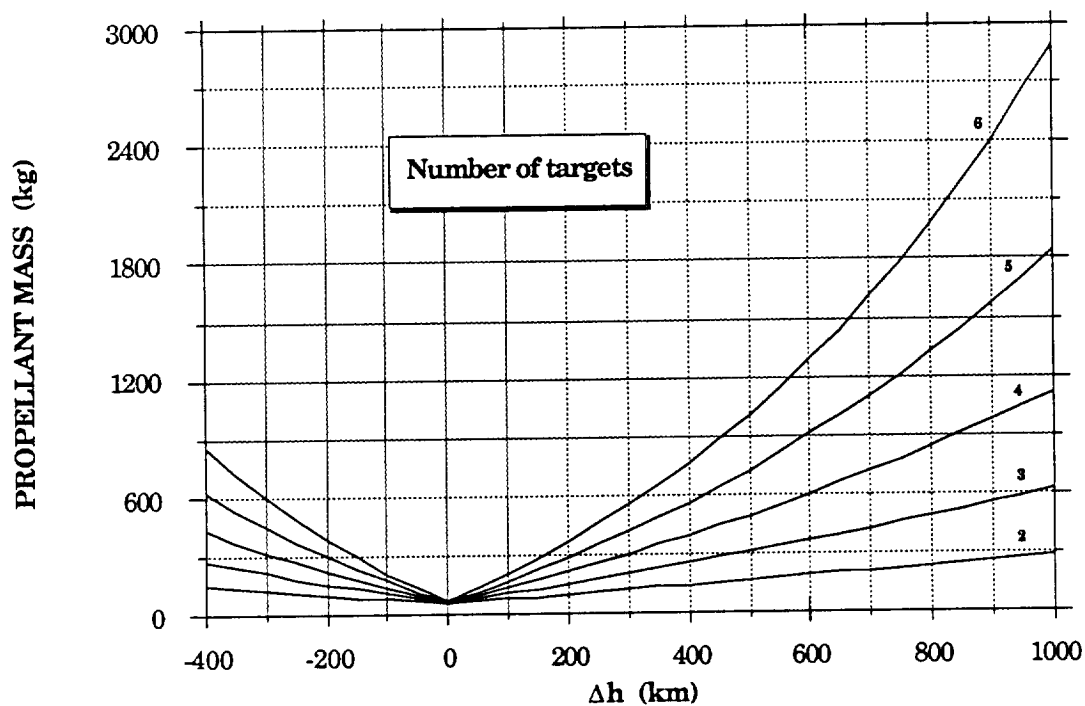


Figure 16. Propellant Mass vs. Δh and number of targets for SRM deorbit mission

The average altitude of the four pieces is 754 km and the average mass is 1956 kg. As mentioned, the expected propellant mass required for this mission will be roughly 1500 kg if a minimum loiter orbit is used. If the ASPOD satellite begins the mission by retrieving the OAO-2 telescope first (a west to east sequence), the

	Perigee (km)	Apogee (km)	i (deg.)	ω (deg.)	Ω (deg.)	Mass (kg)
OAO-2 rocket	7089	7189	35	239	160	1815
OAO-3 rocket	7067	7152	35	350	121	1815
OAO-3	7108	7116	35	293	10	2220
OAO-2	7137	7150	35	315	0	2012

Table 1. Orbital Parameters of Baseline ASPOD Targets (Reference 14).

minimum loiter orbit will be 950 km to allow the remaining pieces of debris to drift towards ASPOD. The total ΔV for this mission is 860 m/s, with a propellant requirement of 1592 kg, based on a specific impulse of 325 seconds. If the mission begins with the OAO-3 rocket (east to west), the minimum loiter orbit will be 550 km allowing ASPOD to drift towards the remaining debris. The total ΔV for this mission is 958 m/s, with a propellant requirement of 1646 kg. The corresponding propellant masses to deorbit the four pieces of debris for the two sequences are 219 kg and 241 kg, respectively. The total longitudinal difference

between the four pieces is 160° , thus, the loiter time to accomplish the transfers is 320 days. Allowing time for the solar cutter to cut the debris into manageable pieces and storing them on board, or, for SRM attachment, the total mission time will be approximately one year.

Another possible mission scenario is a combination of the above approaches. That is, retrieving the telescopes for any valuable materials, but deorbiting the spent booster stages. The propellant necessary to deorbit the boosters first and continue on to the telescopes for retrieval is approximately 652 kg. Where the SRM's for retro firing added a total of 140 kg to the initial mass.

As mentioned before, a large quantity of debris is located at inclinations of 65° , 74° and 82° , with varying altitudes. The use of nodal regression is a convenient method for retrieving debris with various altitudes. The most economical mission sequence is to rendezvous with the piece of debris that is at the highest altitude and work back down to the piece of debris in the lowest altitude. As the mission begins at the debris with the highest altitude, ASPOD is confined to a west to east sequence. Therefore, the mission may be on the order of 2-3 years. Since most of the debris in the 600 km to 1500 km altitude range have lifetimes of 100 years or more, a 2 or 3 year mission is not an inhibiting circumstance.

It is clear from the above analysis that the most fuel efficient method for removing orbital debris is to attach an SRM to the debris and send it into an orbit that will quickly decay and burn up in the atmosphere. This is not to say that it is the "best" method. The technical aspects of attaching the SRM's and assuring that the debris is properly deorbited presents formidable difficulties. The opportunity will also be missed to take advantage of the material resources that the debris population presents. Many, thousands of kilograms of high grade, "space-age" material may be salvaged by the abilities of the ASPOD spacecraft to cut up the debris and store it onboard for eventual recovery by the Space Shuttle, or a controlled splash-down. Thus, the ASPOD spacecraft has two potential mission purposes; 1.) to clean up the increasingly polluted LEO region and 2.) to retrieve useful materials in orbit about the Earth for possible recycle or long duration exposure study.

Proximity Operations 15

This analysis will bring ASPOD within 50m of the orbiting debris. A local coordinate system is defined and two approaching techniques are briefly explained.

The coordinate system to used has its origin at the center of mass on the respective orbital vehicle. This system is the local vertical/local horizontal (LVLH). The x-axis (X_{LVLH}) lies along the velocity vector of the orbital vehicle, while the z-axis (Z_{LVLH}) lies along the radius vector between the earth and the orbiting vehicle. The y-axis (out of plane) completes the coordinate system (Figure 17)

Two approach trajectories ASPOD can use are:

1. V-bar
2. R-bar.

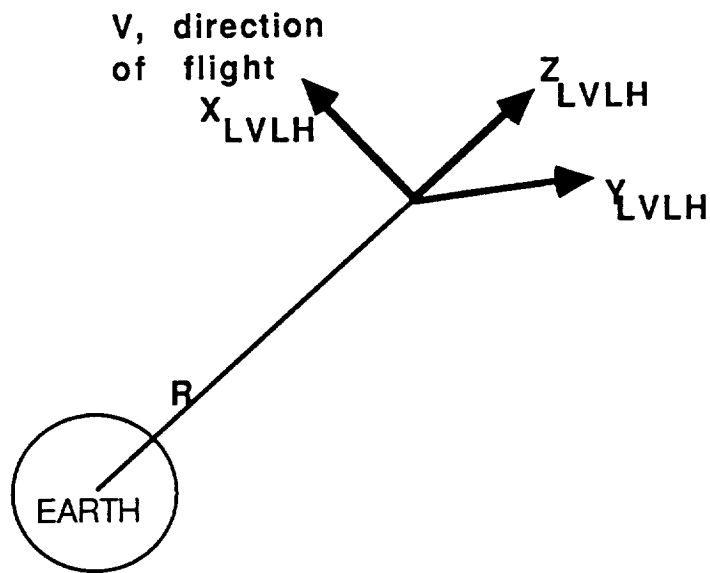


Figure 17 Local Vertical, Local Horizontal Coordinate System

For V-bar, the ASPOD approaches the debris along X_{LVLH} axis. The advantage of this method is low fuel requirements for station keeping. However, since ASPOD would never "catch" the debris while along the same orbit, there is added fuel costs for "accelerating" ASPOD, then braking once the debris is reached. This has an additional side effect of plume impingement which could potentially tumble the debris.

The R-bar method uses an approach along the Z_{LVLH} axis. This approach is still an in plane approach, but the orbit of ASPOD will be slightly greater than or less than the debris. One advantage of this method is plume impingement on the debris while station keeping is minimized. There will, however, be some station keeping required since velocities of the two orbits will be slightly different at any given time.

As previously discussed, nodal regression and Hohmann transfer techniques will be used for matching orbits. R-bar approaches are recommended for final approach to minimize plume impingement effects.

Once it has been determined that ASPOD is in the proper plane in order to rendezvous with the debris, ASPOD will fire its engines so that it may change to an orbit slightly larger than that of the debris. This new orbit should be less than 10km higher than the debris so that ASPOD's microwave radar can locate the debris. The rendezvous sequence is summarized below in figure 18.

When ASPOD has achieved this orbit, it determines relative velocities and range between itself and the debris using Doppler effect and its microwave radar. If the orbit is on the outer limit of radar range (10km), the orbit will be quickly adjusted so that ASPOD can be brought to within 1km of the debris. One principal reason for doing this is the relatively large amount of fuel that would be needed for station keeping with this altitude difference.

Assuming the debris to be in a circular orbit at 700km, and ASPOD's initial orbit at 710km, velocity differences are approximately 5.3 m/s. This is a worse case scenario since the orbits are somewhat higher than this, therefore velocity differences are lower. The 10km upper limit is chosen because it is

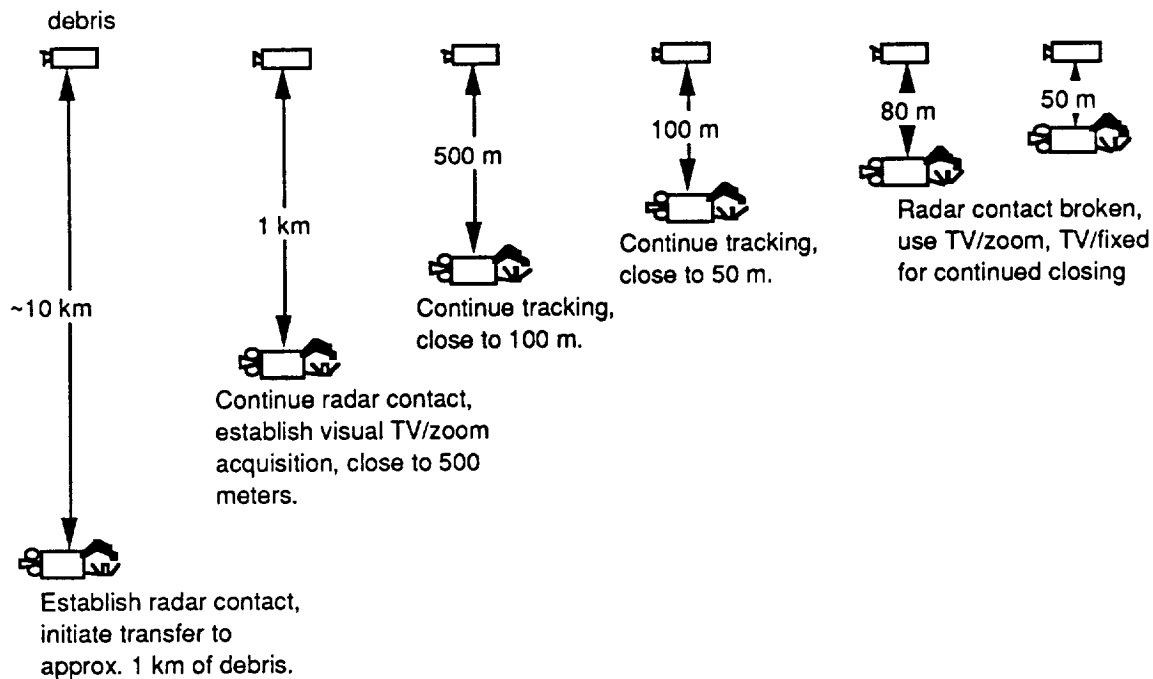


Figure 18. ASPOD Proximity Operations Summary

the outer range of the microwave radar. It is probable that the initial orbit will be closer than 10km, but no closer than 1km. In any case, once orbit, and radar contact is established, ASPOD will quickly fire its engines to come within 1km. ASPOD will constantly be monitoring relative ranges and velocities between itself and the debris.

Once 1km is reached, the TV/ZOOM lens can be used to locate the debris. At this point station keeping would require firing the engines for an approximate velocity deficit of .53 m/s. The ASPOD can now determine orbital attitude of the debris and recheck range and velocity differences. Now ASPOD can begin to slowly move closer to the debris.

The ASPOD uses the onboard computer to analyze range and velocity then fires its engines to orbit within 500m of the debris. During this time, ASPOD is constantly being updated on the relative velocities and range between itself and the debris. Any deviation that could be considered catastrophic can be corrected by firing the appropriate engine to compensate. Again ASPOD uses the radar to measure ranges and velocities while ground control can use the TV/ZOOM lens to observe the mission if required.

These same procedures are repeated to bring the orbiter to within 100m, then 50m of the debris. Each step is carefully analyzed and as the orbiter approaches the debris in each step, greater care and precision are required so as not to de-stabilize the debris any further than it may already be. Once inside approximately 80m the microwave radar is essentially useless, and a TV/FIXED lens can be used in conjunction with the TV/ZOOM lens.

At 50m above the debris, station keeping for ASPOD consists of an approximate velocity deficit of .03 m/s in X_{LVLH} axis. It may be decided to de-spin or de-tumble the debris from this range, or closer proximity may be needed in order to complete processing of the debris.

Once the debris is processed, ASPOD continues on its mission and the cycle is repeated. For nodal regression, ASPOD moves to a higher (or lower) orbit, then approaches the debris, in plane, using R-bar method.

For the first piece of debris using a launch from the Space Shuttle or ELV, the same R-bar approach may be used except ASPOD approaches from a lower orbit (higher velocity) using similar techniques.

Whether or not a closer approach is needed to de-spin or de-tumble the debris is unknown at this point. An in depth study is being conducted at this point to develop a method to de-spin and make contact with the debris.

Spacecraft Configuration

The ASPOD spacecraft is currently envisioned as consisting of a main cylindrical body section approximately five meters in length by three meters in diameter. Attached to the aft end of this central body are a main rocket motor, solar arrays, attitude thrusters and tankage. A solar cutter device, two robot arms, and most spacecraft instrumentation are attached to the front of the spacecraft. Within the central body section is cargo space for the storage of orbital debris or other mission hardware, such as solid rocket motors for the deorbiting of debris not to be retrieved within ASPOD itself. The basic configuration for the ASPOD spacecraft is depicted in Figure 19.

For the mission scenario investigated, the initial (wet) mass of the ASPOD spacecraft is approximately 2500 kg, with a dry mass of approximately 560 kilograms. The overall spacecraft mass is dominated by propellants, which account for the majority of the initial total (Table 2).

The ASPOD system has been logically divided into ten major subsystems, namely: Structures, Electrical Power, Thermal Control, Propulsion, Attitude Control, Communications, Command & Data Handling, Robotics, Solar Cutter, and Guidance & Navigation. These main subsystems are depicted in block diagram format, in Figure 20. The conceptual breakout of a spacecraft configuration is somewhat subjective as many functions overlap. The following sections describe the major ASPOD subsystems in greater detail. The overall ASPOD configuration and mass estimates are based on comparison with various other spacecraft and on technological forecasts contained in references 16-21.

Structure - The main ASPOD structure includes an approximately 300 kilogram main body with robotic arms, a solar cutter, and various sensors and spacecraft equipment attached to the front end. Within this main body is space for the storage of any orbital debris for which retrieval is desired. The front of the spacecraft body constitutes the main spacecraft bus and must provide mounting and power connections for various spacecraft sensors and instruments for navigation and debris processing purposes. The Propulsion, Attitude Control, and Electrical Power subsystems are mounted to the rear of the ASPOD main body. Mounts and launch vehicle interfaces must also be provided here.

Electrical Power - The average electrical load for ASPOD is estimated at approximately 50 watts. Primary power is supplied to ASPOD from solar arrays deployed on booms at the aft end of the spacecraft. Power for peak activities and during periods of eclipse is provided by nickel-hydrogen type rechargeable batteries, massing approximately 5 kilograms. The amount of

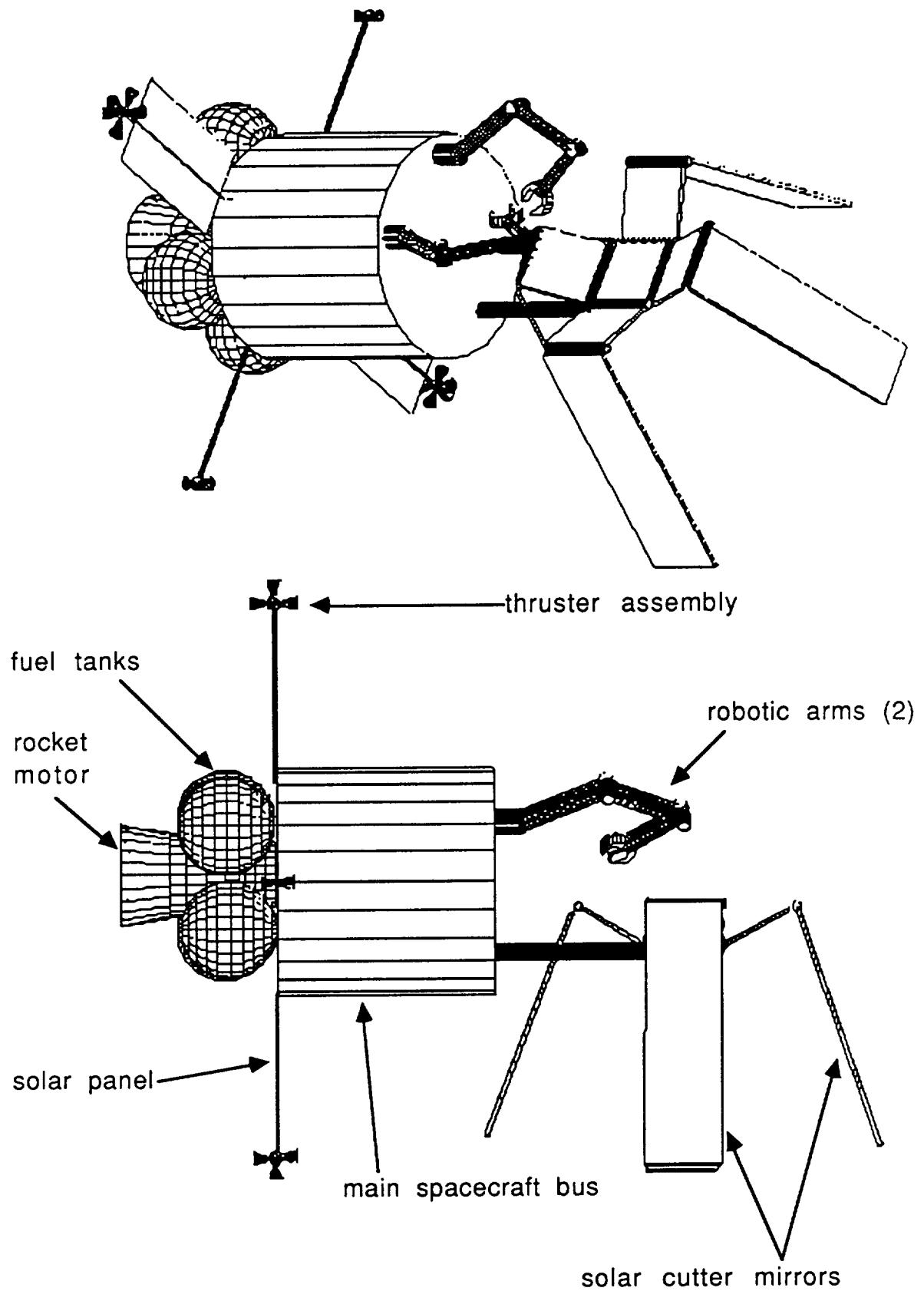


Figure 19. ASPOD Configuration

ASPOD BLOCK DIAGRAM

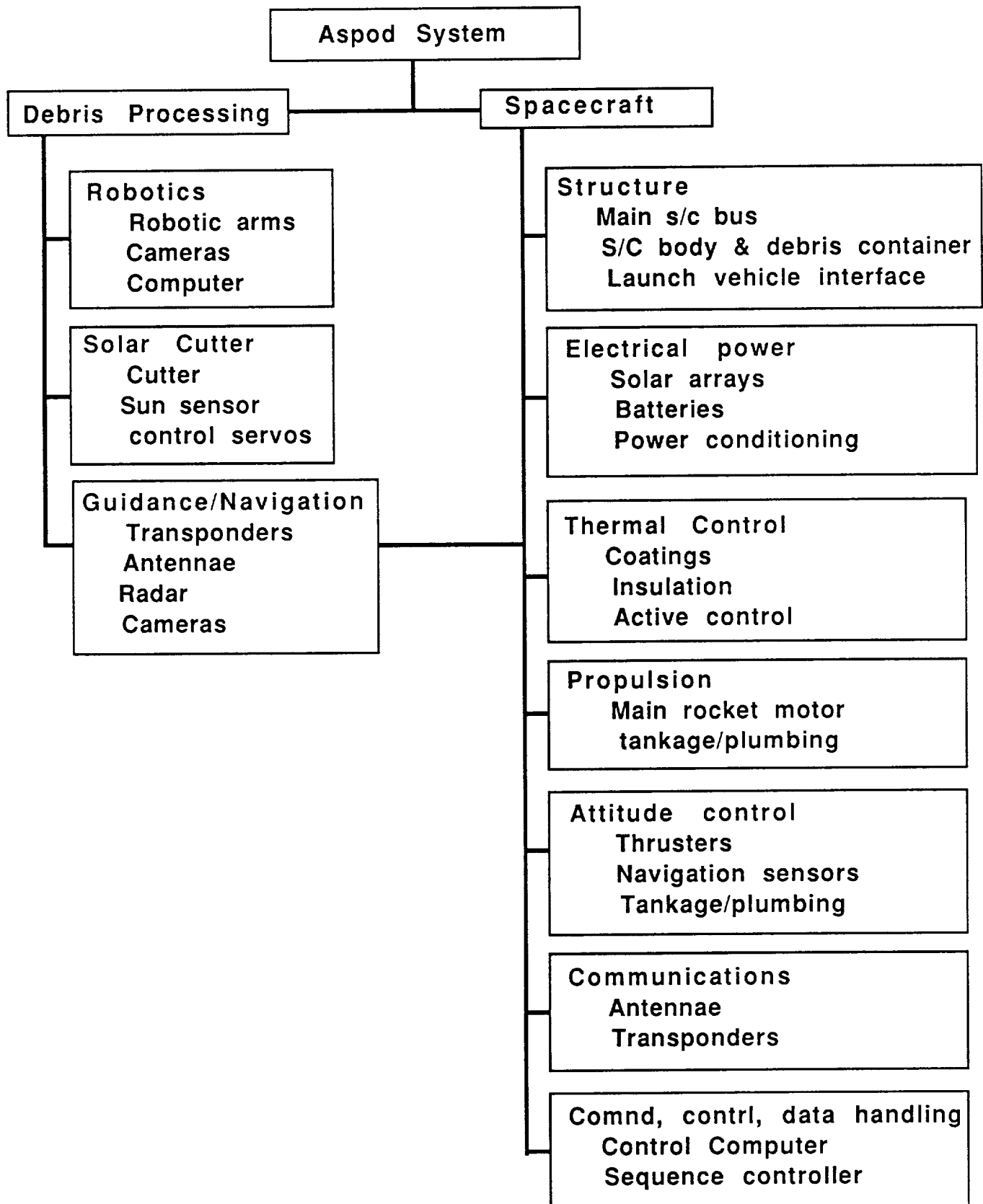


Figure 20. ASPOD System Block Diagram

Table 2. ASPOD Mass Breakout

SYSTEM/COMPONENT	MASS, kg	COMMENTS
STRUCTURES & MECHANISMS		
S/C systems bus	30	estimate
Main Spacecraft Structure	300	parametric
Launch Vehicle interface	15	estimate
Miscellaneous mechanisms	5	estimate
ELECTRICAL POWER		
Solar arrays	2	base on power budget and battery charging needs
Batteries	5	Nickel Hydrogen type rechargeable
Power Conditioning	3	based on Galileo Probe
Cabling	1.4	Galileo
Battery monitoring harness	0.9	est
Miscellaneous	1	est
THERMAL CONTROL		
total	12	parametric estimate
GUIDANCE, NAVIGATION		
Tracking Transponder	1.4	VEGA Model 366C
TT Antenna	1.4	VEGA 106002
Ranging Radar Transponder	1.4	estimate
RR Antenna	.4	estimate
Array Detector Camera	2	estimate
inertial measuring unit	1.8	laser gyro type
Horizon Sensor	1.2	need to periodically reset IMU
Magnetometer	1.2	
accelerometers	0.5	SOA Bell Textron #11
Cabling	0.5	estimate
PROPULSION		
Propellants	1600	ASPOD mission to 4 debris using nodal regression techniques
main rocket motor	8	for near impulsive delta-v based on sc + cargo mass
Tankage	37	2.3% mass fraction of propellant
Plumbing/valving	4	size from other spacecraft
Wiring	0.5	estimate
ATTITUDE CONTROL		
Propellants	0	incl in propulsion
Thrusters	19.6	4 @ 1.4 kg each net hardware
Tankage	0	included in propulsion
Plumbing/Valving	2	estimate
Sensors, filters, heaters	1	(prevent freezing of propellants)
AUTOMATION/ROBOTICS		
Robot arms	20	estimate
Robot TV camera	2	typical
Solar Cutter	40	(size for adequate kerf during solar passage)
Sun sensor	1.2	estimate
Robot Control Computer	5	estimate

Table 2. ASPOD Mass Breakout (continued)

COMMUNICATIONS

Omnidirectional Antennae	1.4	Galileo
Transponders	5	Lifesat
Wiring	0.5	estimate
oscillator/exciter	2.8	estimate

OVERALL SC COMMAND, CONTROL, DATA HANDLING

Control Computer	7	Lifesat
Sequence Controller	1.2	Lifesat
Wiring & Connectors	4	Lifesat
Other electronics	3	estimate

SOLID ROCKET MOTORS

total	284	4 @ 71 kg
-------	-----	-----------

TOTAL: 2447.3

dry s/c only 563.3 (not including SRM's or propellants)

time a spacecraft spends in and out of the sun varies with season, orbital inclination and altitude. For the ASPOD mission scenarios investigated, the average time in the sun was approximately 67 percent, with a worst case of 27 percent. Thus, the solar panels must be sized to provide power in excess the nominal spacecraft load to provide the batteries sufficient charge to operate the spacecraft during periods of darkness. For the spacecraft load of 50 watts, a total collection area of approximately 1.7 square meters are needed for the solar arrays (see Appendix), with a mass of 12 kilograms, (based on expected array performance available by the mid-1990's)²². It may also be desirable to place additional solar cells on the body of the spacecraft itself to allow greater flexibility of orientation during debris processing operations.

Thermal Control - A detailed thermal analysis for ASPOD has not yet been conducted. However, the relatively low nominal power dissipation of ASPOD and the relatively short orbital periods of low earth orbits (compared to geostationary orbits) indicate that passive means of thermal control such as insulation blankets, and absorptive/emissive/reflective coatings should be sufficient for most thermal control. The thermal control system is estimated to mass approximately 12 kilograms. Small electrical heaters will also be needed in choice locations to prevent propellants from freezing and to keep thrusters working properly.

Propulsion - The ASPOD propulsion system is designated as a pressure fed hydrazine/nitrogen tetroxide bipropellant system. The main rocket motor weighs 8 kilograms and provides 2000 N of force with a specific impulse of approximately 325 seconds. Propellants are stored in four pressurized tanks mounted at the aft of the spacecraft. The hydrazine propellant and nitrogen tetroxide oxidizer are storable liquids at room temperatures - hence avoiding the refrigeration power consumption and other difficulties associated with cryogenic (hydrogen/LOX) propellants.

Attitude Control - The attitude control system consists of two sets of five thrusters mounted at the ends of the solar array booms, as well as two additional sets of two thrusters on booms set perpendicular to the solar arrays,

for a total of 14 thrusters massing 20 kilograms (1.4 kg net hardware each). Such thrusters are typified by the Marquardt Model number R-6C (used for the GOES satellite) which provide 22 N of thrust. The thrusters are arranged to provide six degrees of freedom (translation and/or rotation about all three principle axes) which will be necessary for proximity [i.e. to debris] operations. The attitude control thrusters use the same propellants and tankage as the main rocket engine and have a specific impulse of approximately 280 seconds.

Communications - ASPOD will need to telemeter data to the earth and receive commands from the ground. Transponders and antennae used for other spacecraft should be quite adequate for the needs of ASPOD. ASPOD should, however, have omnidirectional (low gain) communications capabilities to allow orientational freedom during debris processing.

Command and Data Handling - The ASPOD spacecraft will be largely autonomous and computerized, with computers controlling spacecraft operations and monitoring the health of various ASPOD systems. Functions will also include pattern recognition capabilities in conjunction with navigation sensors and cameras.

Robotics - ASPOD will be equipped with two robotic arms similar in concept to the space shuttle RMS system but smaller. Two arms are necessary to allow pieces of debris to be dismembered and stored, or to hold the debris with one arm while the other performs other operations, such as the retrieval and mounting of a solid rocket motor. Television cameras will be necessary to monitor the debris processing robotic activities, and the robot arms will be computer controlled.

Solar Cutter - The ASPOD solar cutting system is currently envisioned as consisting of an assembly of gold-plated mirrors (to better reflect infra-red) and fresnel lenses to focus sunlight to cut or weld debris. A sensor will track the sun and orient the cutter assembly for optimal cutting by means of a feedback servo-control system. The solar cutter system is estimated to weigh approximately 40 kilograms.

Guidance/Navigation - The ASPOD spacecraft will be three-axis stabilized with inertial (gyroscopic) guidance referenced to an inertial measuring unit (IMU). Sensors including a horizon sensor, sun sensor, and magnetometer (to measure earth's magnetic field) will be used to reset the inertial measuring unit from any gradual drift. Accelerometers will measure and verify spacecraft velocity changes for orbital maneuvers. Orbital debris will be tracked by onboard radar and camera systems. However, before the debris is in close enough proximity to ASPOD for the use of onboard tracking methods, ground tracking may be necessary. This can be made easier by providing the ASPOD spacecraft with a tracking transponder to facilitate more accurate tracking of ASPOD's position from the ground.

The ASPOD configuration presented above is not finalized and will become more refined as the concepts of an ASPOD mission mature. Still, it provides a good general idea of what sort of spacecraft might carry out the envisioned debris processing mission.

Propulsion and Attitude Control Systems

The propulsion system includes the main thruster, the attitude control system, and also the fuel tanks. The main thruster for the satellite will be a

high performance, pressure fed Hydrazine/Nitrogen tetroxide (MMH/N₂O₄) engine with the specifics presented in Table 3 as shown below.

Table 3 - Main Thruster

Vacuum Thrust	2000 N
Vacuum Isp	325 sec (MMH/N ₂ O ₄)
Mixture Ratio	1.0
Nozzle area ratio	300
Throat Temperature	1400-1600 K
Weight of Engine	8.5 kg
Weight of Tanks	~2% of fuel weight
Weight of Fuel	Will correspond to mission

The engine chosen has been built and tested in a vacuum chamber.²³ This engine and fuel tank system is fully capable of completing many potential ASPOD mission scenarios.

The attitude control system will consist of two sets of five clustered thrusters mounted on the solar array booms, and two additional sets of two thrusters each mounted on booms set perpendicular to the solar arrays to provide six degrees of freedom (see Figure 19). The auxiliary thruster chosen for the ASPOD attitude control system was the Marquardt model R-6C pressure fed, bipropellant thruster. This thruster uses mono-methyl hydrazine as the fuel, and nitrogen tetroxide as the oxidizer. The propellant is the same as that of the main thruster; this is quite beneficial as it is unnecessary to provide tanks specifically for use by the auxiliary propulsion system (i.e., both the main and auxiliary propulsion systems operate off the same tanks). The performance and weight characteristics of these thrusters are shown below in Table 4.

Table 4- Auxiliary Thrusters
(For A Single Thruster)

Vacuum Thrust	22 N
Vacuum Isp	280 sec
Number of Starts	400,000
Minimum Pulse	5 msec
Power (one pair in parallel)	15 Watts @ 28V
Weight of thruster	0.66 kg
Weight of Support Structure	36 kg
Weight of Propellant	Included in main system

The number of starts is defined as the maximum number of firings during the lifetime of the satellite. The Minimum pulse is defined as the minimum time to steady state operation of the thruster. The bipropellant auxiliary thruster was selected out of a number of choices based on thrust, impulse, response, and ease of integration with main propulsion system. A comparison of some types of thrusters is available in the appendix.

The hydrazine and nitrogen tetroxide fuel and oxidizer system has been very popular for use on spacecraft. It can be easily stored at room

temperatures and it gives a fairly high specific impulse. This propellant requires a number of heaters throughout the satellite in order to prevent freezing. The heaters are also necessary to prevent the hydrazine from caking around the inlet tubes, which could potentially cause the thruster to explode.²⁴ The caking is a result of the impulse bits of the thrusters (i.e., thrusters may not fire for a full 5 msec, thus leaving residue).

The propulsion system was chosen to provide high reliability, good inherited characteristics and ease of integration.

Launch Vehicles

The Titan III launch vehicle was baselined to boost ASPOD into orbit; the selection was based on a number of considerations including: performance, weight, and dimensions, and cost. The Titan III is fully capable of lifting several ASPOD's into low Earth orbit. The Titan III provides launch services for both commercial and government payloads to either low Earth orbit, or geosynchronous transfer orbit from launch complex facilities at Cape Canaveral Air Force Station in Florida.²⁵ The space shuttle, Delta, and Ariane launch vehicles would also be feasible candidates to boost ASPOD into orbit.

The configuration of the Titan is a combination of solid and liquid rockets with various choices in payload fairings. The rocket consists of two solid rocket motors (SRM's), and two liquid stage motors.

The performance characteristics of the Titan rocket provide high thrust with capability of launching spacecraft into LEO (low Earth orbit), or GTO (geosynchronous transfer orbit). The thrust for the SRM's is 12,420 kN, and the thrusts for the liquid first and second stages are 2429 kN, and 463 kN respectively. The specific impulses (Isp) for the SRM's are 265 seconds, and the Isp for the liquid first and second stages is 305 seconds.

The process of launching a satellite into LEO takes less than nine minutes. The maximum static acceleration throughout the launch phase does not exceed 4 g's.

The predicted launch weight is approximately 1.53 million pounds with a maximum payload capability of 31 thousand pounds for LEO.

Summary

The Space Engineering Design program has been on track. Seven (plus two) new undergraduate students were involved in the 1988-89 school year; one more has been hired for the summer of 1989. David Campbell continues to be the graduate student in charge of the USRA project; he continues to spend his summers at JPL.

The ASPOD solar cutter, auto sun track mechanisms have all been built and are working satisfactorily. One (gathering) robotic arm has been built and is also operating satisfactorily. The second robotic arm for positioning/assembly of the cut parts is being built. The project has drawn considerable news media attention with many popular and science forums. It is expected that with continued progress a hardware experiment can be conducted on a GETAWAY special in 1991 or 1992.

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APPENDIX A

ORBITAL MANUEVERING ANALYSIS

Computer program for direct transfer calculations.

User must input the following variables:

DRYWT	- ASPOD dry mass in kg (minus fuel tank)
SPCFIMP	- main thruster specific impulse (sec.)
N	- number of debris targets
PARKALT	- initial parking orbit altitude (km)
FINALT	- final parking orbit altitude (km)
PERC	- fuel tank mass percentage
RP(N)	- perigee radius for Nth target (km)
RA(N)	- apogee radius for Nth target (km)
ARG(N)	- argument of perigee for Nth target (deg.)
AI(N)	- inclination of Nth target (deg.)
ZLONG(N)	- longitude of ascending node for Nth target (deg.)
MASS(N)	- mass of Nth target (kg)

Program will output the following:

ΔV 's for each impulse
Total propulsion mass necessary for mission
Initial total system mass for mission

```

C      PROGRAM USRA
C
C *****
C      THIS PROGRAM WILL FIND THE DELTA V'S AND PROPELLANT
C      REQUIREMENTS FOR A VARIETY OF ASPOD MISSION PROFILES.
C *****
C      IMPLICIT DOUBLE PRECISION (A-H,O-Z)
C
C      DIMENSION RP(10),RA(10),AI(10),ARG(10),ZLONG(10),ANOM(10),A(10)
C      DIMENSION ECC(10),P(10),PER(10),DELANOM(10),R1(2),R2(2),V1(2)
C      DIMENSION V2(2),XANOM(10),C(100),YY(100),DV(50),SUM(50)
C      REAL MASS(10)
C      CHARACTER*64 FILENAME
C *****
C      INPUT ROUTINE:  INPUT PROGRAM CONTROL DATA AND ORBITAL PARAMETERS
C *****
C
C      WRITE(*,*) ' '
C      WRITE(*,*) '                               A S P O D '
C      WRITE(*,*) ' '
C      WRITE(*,*) '          AUTONOMOUS SPACE PROCESSOR FOR ORBITAL DEBR
C *IS '
C      WRITE(*,*) '          -----
C *-- '
C      WRITE(*,*) ' '
C      WRITE(*,*) ' '
C      WRITE(*, '(A\)' ) '          - DO YOU WISH TO SAVE THE OUTPUT (Y/N)? '
C      READ(*, '(A)' ) ANSWER
C      IF (ANSWER.EQ.'Y'.OR.ANSWER.EQ.'y') THEN
C          WRITE(*, '(A\)' ) '          - ENTER THE OUTPUT FILE NAME: '
C          READ(*, '(A)' ) FILENAME
C          OPEN (6,FILE=FILENAME,ACCESS='SEQUENTIAL',STATUS='NEW')
C          ICHECK = 1
C      END IF
C      WRITE(*,*) ' '
C      WRITE(*,*) ' '
C      WRITE(*, '(A\)' ) '          1 - WHAT IS THE DRY WEIGHT (Kg)? '
C      READ(*,*) DRYWT
C      WRITE(*, '(A\)' ) '          2 - WHAT IS THE SPECIFIC IMPULSE OF THE TH
C *RUSTER? '
C      READ(*,*) SPCFIMP
C      WRITE(*, '(A\)' ) '          3 - HOW MANY PIECES OF DEBRIS? '
C      READ(*,*) N
C      WRITE(*, '(A\)' ) '          4 - WHAT IS THE INITIAL PARKING ORBIT ALTI
C *TUDE (KM)? '
C      READ(*,*) PARKALT
C      WRITE(*, '(A\)' ) '          5 - WHAT IS THE FINAL PARKING ORBIT ALTITU
C *DE (KM)? '
C      READ(*,*) FINALT
C      WRITE(*, '(A\)' ) '          6 - WHAT IS THE FUEL TANK MASS PERCENTAGE?
C * '
C      READ(*,*) PERC
C
C      WRITE(*,*) ' '

```

```

99  WRITE(*, '(A\)' ) '          WOULD YOU LIKE TO MAKE A CHANGE (Y/N)? '
    READ(*, '(A)' ) YESNO
    IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
      WRITE(*, '(A\)' ) '          WHAT NUMBER IS TO BE CHANGED? '
      READ(*,*) NUM
      IF (NUM.EQ.1) GO TO 501
      IF (NUM.EQ.2) GO TO 502
      IF (NUM.EQ.3) GO TO 503
      IF (NUM.EQ.4) GO TO 504
      IF (NUM.EQ.5) GO TO 505
      IF (NUM.EQ.6) GO TO 506
501  WRITE(*, '(A\)' ) '          1 - WHAT IS THE DRY WEIGHT (Kg)? '
      READ(*,*) DRYWT
      GO TO 99
502  WRITE(*, '(A\)' ) '          2 - WHAT IS THE SPECIFIC IMPULSE OF THE T
      *HRUSTER? '
      READ(*,*) SPCFIMP
      GO TO 99
503  WRITE(*, '(A\)' ) '          3 - HOW MANY PIECES OF DEBRIS? '
      READ(*,*) N
      GO TO 99
504  WRITE(*, '(A\)' ) '          4 - WHAT IS THE INITIAL PARKING ORBIT ALT
      *ITUDE (KM)? '
      READ(*,*) PARKALT
      GO TO 99
505  WRITE(*, '(A\)' ) '          5 - WHAT IS THE FINAL PARKING ORBIT ALTIT
      *UDE (KM)? '
      READ(*,*) FINALT
      GO TO 99
506  WRITE(*, '(A\)' ) '          6 - WHAT IS THE FUEL TANK MASS PERCENTAGE
      *? '
      READ(*,*) PERC
      GO TO 99
    END IF
    WRITE(*,*) ' '
    WRITE(*,*) ' '
    WRITE(*,*) ' *** INPUT THE ORBITAL PARAMETERS IN THE ORDER OF REND
    *EZVOUS *** '

C
    DO 7 L = 1,N
      WRITE(*,*) ' '
      WRITE(*,1000)L
1000  FORMAT(5X,'TARGET #',I2/)
      WRITE(*, '(A\)' ) '          1 - WHAT IS THE PERIGEE RADIUS (KM)? '
      READ(*,*) RP(L)
      WRITE(*, '(A\)' ) '          2 - WHAT IS THE APOGEE RADIUS (KM)? '
      READ(*,*) RA(L)
      WRITE(*, '(A\)' ) '          3 - WHAT IS THE ARGUMENT OF PERIGEE (DEG)
      *? '
      READ(*,*) ARG(L)
      WRITE(*, '(A\)' ) '          4 - WHAT IS THE INCLINATION (DEG)? '
      READ(*,*) AI(L)
      WRITE(*, '(A\)' ) '          5 - WHAT IS THE RAAN (DEG)? '
      READ(*,*) ZLONG(L)
      WRITE(*, '(A\)' ) '          6 - WHAT IS THE MASS (Kg)? '
      READ(*,*) MASS(L)
      WRITE(*,*) ' '
88    WRITE(*, '(A\)' ) '          WOULD YOU LIKE TO MAKE A CHANGE (Y/N)? '
      READ(*, '(A)' ) ANS
      IF (ANS.EQ.'Y'.OR.ANS.EQ.'y') THEN

```

```

WRITE(*,'(A\)' ) '          WHAT NUMBER IS TO BE CHANGED? '
READ(*,*) NNN
IF (NNN.EQ.1) GO TO 1
IF (NNN.EQ.2) GO TO 2
IF (NNN.EQ.3) GO TO 3
IF (NNN.EQ.4) GO TO 4
IF (NNN.EQ.5) GO TO 5
IF (NNN.EQ.6) GO TO 6
1  WRITE(*,'(A\)' ) '          1 - WHAT IS THE PERIGEE RADIUS (KM)? '
   READ(*,*) RP(L)
   GO TO 88
2  WRITE(*,'(A\)' ) '          2 - WHAT IS THE APOGEE RADIUS (KM)? '
   READ(*,*) RA(L)
   GO TO 88
3  WRITE(*,'(A\)' ) '          3 - WHAT IS THE ARGUMENT OF PERIGEE (DEG
*)? '
   READ(*,*) ARG(L)
   GO TO 88
4  WRITE(*,'(A\)' ) '          4 - WHAT IS THE INCLINATION (DEG)? '
   READ(*,*) AI(L)
   GO TO 88
5  WRITE(*,'(A\)' ) '          5 - WHAT IS THE RAAN (DEG)? '
   READ(*,*) ZLONG(L)
   GO TO 88
6  WRITE(*,'(A\)' ) '          6 - WHAT IS THE MASS (Kg)? '
   READ(*,*) MASS(L)
   GO TO 88
END IF
7  CONTINUE
C
C *****
C  DEFINE CONSTANTS, WRITE DATA TO LU6, CHANGE ALL ANGLES TO RADIANS
C *****
C
C  PI = 4.0*ATAN(1.0)
C  ZMU = 398600.0
C  ALPHA = -1000.0/(SPCFIMP*9.81)
C  PERC = PERC/100.
C
C  IF (ICHECK.EQ.1) THEN
C    WRITE(6,899)
C    WRITE(6,900)
C  END IF
899  FORMAT(25X,24HASPOD MISSION PARAMETERS)
900  WRITE(*,900)
   FORMAT(/5X,6HTARGET,3X,7HPERIGEE,3X,6HAPOGEE,4X,5HINCL.,5X,4HRAAN,
*      5X,4HARG.,5X,4HMASS/5X,62H-----/
*-----/)
C
DO 8 L = 1,N
  IF (ICHECK.EQ.1) THEN
    WRITE(6,901)L,RP(L),RA(L),AI(L),ZLONG(L),ARG(L),MASS(L)
  END IF
  WRITE(*,901)L,RP(L),RA(L),AI(L),ZLONG(L),ARG(L),MASS(L)
901  FORMAT(7X,I2,5X,F7.1,2X,F7.1,4X,F5.1,4X,F5.1,5X,F5.1,4X,F6.1/)
  A(L) = (RP(L)+RA(L))/2.0
  ECC(L) = 1.0-RP(L)/A(L)

```

```

      P(L) = A(L)*(1.0-ECC(L)**2)
      AI(L) = AI(L)*PI/180.0
      ARG(L) = ARG(L)*PI/180.0
      ZLONG(L) = ZLONG(L)*PI/180.0
8      CONTINUE
C
      IF (ICHECK.EQ.1) THEN
        WRITE(6,902) DRYWT,SPCFIMP,PARKALT,FINALT
      END IF
      WRITE(*,902) DRYWT,SPCFIMP,PARKALT,FINALT
902    FORMAT(/5X,19HASPOD DRY WEIGHT = ,F6.2,3H Kg,/5X,19HSPECIFIC IMPU
      *LSE = ,F6.2,5H SECS,/5X,25HPARKING ORBIT ALTITUDE = ,F7.2,3H KM,
      */5X,23HFINAL ORBIT ALTITUDE = ,F7.2,3H KM)
C
      IF (ICHECK.EQ.1) THEN
        WRITE(6,903)
        WRITE(6,904)
      END IF
      WRITE(*,903)
903    FORMAT(/27X,21HASPOD MISSION PROFILE/)
      WRITE(*,904)
904    FORMAT(15X,3HDV1,7X,3HDV2,7X,3HPHI,7X,3HPSI,6X,5HANOM1,5X,5HANOM2
      *,/15X,54H-----)
C
C *****
C
C      CALCULATIONS FOR TRANSFER FROM PARKING ORBIT
C
C *****
C
C      PERI = 6372+PARKALT
      APO = RA(1)
      SEMIAX = PERI+APO
      VTRAN1 = DSQRT(2.0*ZMU*(1.0/PERI-1.0/SEMIAX))
      VPER = DSQRT(ZMU/PERI)
      DV(1) = ABS(VTRAN1-VPER)
      VTRAN2 = DSQRT(2.0*ZMU*(1.0/APO-1.0/SEMIAX))
      VAPO = DSQRT(2.0*ZMU*(1.0/APO-1.0/(2.0*A(1))))
      DV(2) = ABS(VTRAN2-VAPO)
      IF (ICHECK.EQ.1) THEN
        WRITE(6,905) DV(1),DV(2)
      END IF
      WRITE(*,905) DV(1),DV(2)
905    FORMAT(/5X,4HPARK,3X,F7.4,3X,F7.4,7X,2H.0,8X,2H.09X,2H.0,5X,5H180.
      *0)
C
C *****
C
C      CALCULATIONS FOR TRANSFERS BETWEEN PIECES OF DEBRIS
C
C *****
C
      DO 10 I = 1,N-1
        JKL = 2*I+1
        DELOMEG0 = ZLONG(I)-ZLONG(I+1)
        DELOMEG1 = ABS(DELOMEG0)
        BETA = ACOS(COS(AI(I))*COS(AI(I+1))+COS(DELOMEG1)*SIN(AI(I))*SIN(
      *AI(I+1)))
        ANOM1 = ACOS((COS(AI(I))*COS(BETA)-COS(AI(I+1)))/(SIN(BETA)*SIN(A
      *I(I))))

```

```

ANOM2 = ACOS((COS(AI(I))-COS(AI(I+1))*COS(BETA))/(SIN(BETA)*SIN(A
*I(I+1))))
IF (DELOMEGO.GT.0.0) THEN
  DELANOM(I) = ANOM1
  DELANOM(I+1) = ANOM2
ELSE
  DELANOM(I) = ANOM2
  DELANOM(I+1) = ANOM1
END IF
CHECK1 = ARG(I)-DELANOM(I)
CHECK2 = ARG(I+1)-DELANOM(I+1)
IF (CHECK1.LT.0.0) THEN
  ANOM(I) = ABS(CHECK1)
ELSE
  ANOM(I) = 2*PI-CHECK1
END IF
IF (CHECK2.LT.0.0) THEN
  ANOM(I+1) = ABS(CHECK2)
ELSE
  ANOM(I+1) = 2*PI-CHECK2
END IF
R1(1) = P(I)/(1.0+ECC(I)*COS(ANOM(I)))
R2(1) = P(I+1)/(1.0+ECC(I+1)*COS(ANOM(I+1)))
V1(1) = DSQRT(2.*ZMU*(1.0/R1(1)-1.0/(2.*A(I))))
V2(1) = DSQRT(2.*ZMU*(1.0/R2(1)-1.0/(2.*A(I+1))))
XANOM(I) = ANOM(I)+PI
XANOM(I+1) = ANOM(I+1)+PI
IF (XANOM(I).GT.2*PI) THEN
  XANOM(I) = XANOM(I)-2.*PI
END IF
IF (XANOM(I+1).GT.2*PI) THEN
  XANOM(I+1) = XANOM(I+1)-2.*PI
END IF
R1(2) = P(I)/(1.0+ECC(I)*COS(XANOM(I)))
R2(2) = P(I+1)/(1.0+ECC(I+1)*COS(XANOM(I+1)))
V1(2) = DSQRT(2.0*ZMU*(1.0/R1(2)-1.0/(2.*A(I))))
V2(2) = DSQRT(2.0*ZMU*(1.0/R2(2)-1.0/(2.*A(I+1))))

C
AA = R1(1)+R2(2)
VT1 = DSQRT(2.*ZMU*(1.0/R1(1)-1.0/AA))
VT2 = DSQRT(2.*ZMU*(1.0/R2(2)-1.0/AA))
BETA1 = BETA
A1 = V1(1)**2+VT1**2
A2 = V2(2)**2+VT2**2
B1 = -2.0*V1(1)*VT1
B2 = -2.0*V2(2)*VT2
DV1 = DSQRT(A1+B1*COS(BETA1))
DV2 = DSQRT(A2+B2*COS(BETA-BETA1))
PROPRAT = 1.0-EXP(ALPHA*(DV1+DV2))
DV3 = DSQRT(A1+B1*COS(BETA-BETA1))
DV4 = DSQRT(A2+B2*COS(BETA1))
PROPRAT1 = 1.0-EXP(ALPHA*(DV3+DV4))
IF (PROPRAT1.LT.PROPRAT) THEN
  DV1 = DV3
  DV2 = DV4
  PHI1 = 0.0
  PSI1 = BETA
  PROPRAT = PROPRAT1
ELSE
  PHI1 = BETA

```

```

    PSI1 = 0.0
END IF
AA = R1(2)+R2(1)
VT1 = DSQRT(2.*ZMU*(1.0/R1(2)-1.0/AA))
VT2 = DSQRT(2.*ZMU*(1.0/R2(1)-1.0/AA))
A1 = V1(2)**2+VT1**2
A2 = V2(1)**2+VT2**2
B1 = -2.*V1(2)*VT1
B2 = -2.*V2(1)*VT2
DV11 = DSQRT(A1+B1*COS(BETA1))
DV22 = DSQRT(A2+B2*COS(BETA-BETA1))
PROPRAT2 = 1.0-EXP(ALPHA*(DV11+DV22))
DV33 = DSQRT(A1+B1*COS(BETA-BETA1))
DV44 = DSQRT(A2+B2*COS(BETA1))
PROPRAT3 = 1.0-EXP(ALPHA*(DV33+DV44))
IF (PROPRAT3.LT.PROPRAT2) THEN
    DV11 = DV33
    DV22 = DV44
    PHI2 = 0.0
    PSI2 = BETA
    PROPRAT2 = PROPRAT3
ELSE
    PHI2 = BETA
    PSI2 = 0.0
END IF
IF (PROPRAT2.LT.PROPRAT) THEN
    PHI = PHI2*180./PI
    PSI = PSI2*180./PI
    DV(JKL) = DV11
    DV(JKL+1) = DV22
    ANOM(I) = XANOM(I)
ELSE
    PHI = PHI1*180./PI
    PSI = PSI1*180./PI
    DV(JKL) = DV1
    DV(JKL+1) = DV2
    ANOM(I+1) = XANOM(I+1)
END IF
TRU1 = ANOM(I)*180./PI
TRU2 = ANOM(I+1)*180./PI
IF (ICHECK.EQ.1) THEN
    WRITE(6,906) I, DV(JKL), DV(JKL+1), PHI, PSI, TRU1, TRU2
END IF
WRITE(*,906) I, DV(JKL), DV(JKL+1), PHI, PSI, TRU1, TRU2
906  FORMAT(/6X,1H#,I2,3X,F7.4,3X,F7.4,4X,F5.1,5X,F5.1,6X,F5.1,5X,F5.1
*)
10  CONTINUE
C
C *****
C
C  CALCULATIONS FOR TRANSFER TO FINAL PARKING ORBIT
C
C *****
C
M = 2*N+1
PERIG = FINALT+6372.
AXIS = RA(N)+PERIG
VA1 = DSQRT(2.*ZMU*(1.0/RA(N)-1.0/AXIS))
VA0 = DSQRT(2.*ZMU*(1.0/RA(N)-1.0/(2.*A(N))))
DV(M) = ABS(VA1-VA0)

```



```

VP1 = DSQRT(2.*ZMU*(1.0/PERIG-1.0/AXIS))
VP0 = DSQRT(ZMU/PERIG)
DV(M+1) = ABS(VP1-VP0)

C
IF (ICHECK.EQ.1) THEN
  WRITE(6,907)N,DV(M),DV(M+1)
END IF
WRITE(*,907)N,DV(M),DV(M+1)
907  FORMAT(/6X,1H#,I2,3X,F7.4,3X,F7.4,7X,2H.0,8X,2H.0,6X,5H180.0,8X,
*2H.0)

C
C *****
C
C   CALCULATE THE REQUIRED PROPELLANT MASS, FUEL TANK MASS, AND
C   TOTAL LAUNCH MASS.
C *****
C
DO 21 I = 1,N+1
  MM = 2*I-1
  KK = 2*(N+1)
  SUM(MM) = 0.0
  DO 20 J = MM, KK
    SUM(MM) = SUM(MM)+DV(J)
20  CONTINUE
  C(MM) = 1.0-EXP(ALPHA*SUM(MM))
21  CONTINUE
  PAYLD = 0.0
  DO 30 JJ = 1,N
    II = 2*JJ+1
    PAYLD = PAYLD+C(II)*MASS(JJ)
30  CONTINUE
C
PROPEL = (C(1)*DRYWT+PAYLD)/(1.0-(1.0+PERC)*C(1))
WEIGHT = (1.+PERC)*PROPEL+DRYWT
TANK = PERC*PROPEL

C
IF (ICHECK.EQ.1) THEN
  WRITE(6,908)PROPEL,WEIGHT,TANK
END IF
WRITE(*,908)PROPEL,WEIGHT,TANK
908  FORMAT(/5X,'PROPELLANT MASS = ',F9.2,' Kg',/5X,'LAUNCH MASS = ',F
*9.2,' Kg',/5X,'TANK MASS = ',F7.2,' Kg')

C
WRITE(*,*) ' '
STOP '      ASPOD ANALYSIS COMPLETED '
END

```

SAMPLE CALCULATIONS FOR DIRECT TRANSFER METHOD - 2 TARGET RETRIEVAL.

TRANSFER FROM 500 KM PARKING ORBIT TO FIRST RENDEZVOUS (750 KM - SAME PLANE).

$$V_1 = \sqrt{\frac{\mu}{r}} \quad \mu = 3.986 \times 10^5 \text{ km}^3/\text{s}^2$$
$$r = 500 + 6372 = 6872 \text{ km}$$

$$\therefore V_1 = 76160 \text{ km/s}$$

$$V_2 = \sqrt{\frac{\mu}{r_2}} \quad r_2 = 750 + 6372 = 7122 \text{ km}$$

$$\therefore V_2 = 74811 \text{ km/s}$$

TRANSFER ORBIT : $2a = 6872 + 7122 = 13994 \text{ km}$

$$V_{t1} = \sqrt{2\mu \left[\frac{1}{r_1} - \frac{1}{2a} \right]}$$

$$\therefore V_{t1} = 7.6837 \text{ km/s}$$

$$V_{t2} = \sqrt{2\mu \left[\frac{1}{r_2} - \frac{1}{2a} \right]}$$

$$\therefore V_{t2} = 7.4140$$

$$\text{THUS, } \Delta V_1 = V_{t1} - V_1 = 7.6837 - 7.6160$$

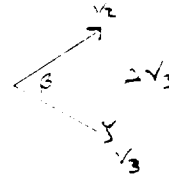
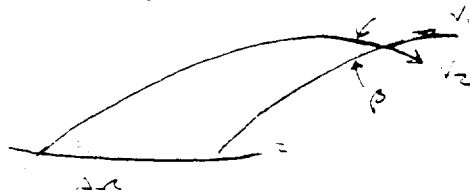
$$\underline{\Delta V_1 = 0.0677 \text{ km/s}}$$

$$\Delta V_2 = V_2 - V_{t2} = 7.4811 - 7.4140$$

$$\underline{\Delta V_2 = 0.0671 \text{ km/s}}$$

ORIGINAL PAGE IS
OF POOR QUALITY

TRANSFER FROM FIRST DEBRIS ORBIT TO
SECOND DEBRIS ORBIT.



$$\cos \beta = -\cos \theta_1 \cos (\pi - \theta_2) + \sin \theta_1 \sin (\pi - \theta_2) \cos 2R$$

FOR THIS EXAMPLE $\theta_1 = \theta_2 = 35^\circ$ $2R = 30^\circ$

$$V_2 = V_3 = 7.4811 \text{ KM/S}$$

WITH $\theta = 35^\circ$ & $2R = 30^\circ$

$$\beta = 17.07^\circ$$

$$\Delta V_3 = \sqrt{V_2^2 + V_3^2 - 2V_2V_3 \cos \beta} \quad (\text{LAW OF COSINES})$$

SIMPLIFY,

$$\Delta V_3 = V_2 \sqrt{2(1 - \cos \beta)}$$

$$\therefore \Delta V_3 = 2.2211 \text{ KM/S}$$

TRANSFER TO FINAL PARKING ORBIT ($h = 500 \text{ KM}$)

$$\Delta V_4 = \Delta V_2 = 0.0671 \text{ KM/S}$$

$$\Delta V_5 = \Delta V_2 = 0.0677$$

MASS CALCULATIONS : $I_{sp} = 325 \text{ SEC}$ $g = 9.81 \text{ M/S}^2$

$$M_{DRY} = 526 \text{ KG}$$

$$\text{TOTAL } \Delta V (\Delta V_T) = 2.4907 \text{ KM/S}$$

TOTAL ΔV AFTER FIRST RENDEZVOUS (ΔV_{T1})

$$\Delta V_{T1} = 2.3559 \text{ KM/S}$$

TOTAL ΔV AFTER SECOND Rendezvous (ΔV_{T2})

$$\Delta V_{T2} = 0.1342 \text{ km/s}$$

MASS EQ. $\frac{M_p}{M_i} = 1 - e^{-\frac{\Delta V}{\Delta V_{sp}}}$ (GENERAL)

FOR THIS MISSION MASS EQ RESULTS IN

$$M_p = C_1 (M_{DRY} - 1.023 M_p) - (C_2 + C_3) M_{DEBRIS}$$

$$M_p = \frac{C_1 M_{DRY} - (C_2 + C_3) M_{DEBRIS}}{1 - 1.023 C_1}$$

WHERE

$$C_1 = 1 - \exp\left[-\frac{\Delta V_{T1}}{\Delta V_{sp}}\right] = 0.512$$

$$C_2 = 1 - \exp\left[-\frac{\Delta V_{T1}}{\Delta V_{sp}}\right] = 0.5224$$

$$C_3 = 1 - \exp\left[-\frac{\Delta V_{T2}}{\Delta V_{sp}}\right] = 0.0414$$

$$M_{DEBRIS} = 2000 \text{ kg}$$

THUS

$$\underline{M_p = 3171.5 \text{ kg}}$$

Computer program for planar transfer calculations.

User must input the following variables:

DRYWT	- ASPOD dry mass in kg (minus fuel tank)
SPCFIMP	- main thruster specific impulse (sec.)
N	- number of debris targets
PARKALT	- initial parking orbit altitude (km)
FINALT	- final parking orbit altitude (km)
RLOIT	- loiter orbit altitude (km)
PERC	- fuel tank mass percentage
RP(N)	- perigee radius for Nth target (km)
RA(N)	- apogee radius for Nth target (km)
MASS(N)	- mass of Nth target (kg)

Program will output the following:

ΔV 's for each impulse
Total propulsion mass necessary for mission
Initial total system mass for mission

PROGRAM NODE

```

C *****
C
C THIS PROGRAM WILL FIND THE DELTA V'S AND PROPELLANT
C REQUIREMENTS FOR A VARIETY OF ASPOD MISSION PROFILES.
C
C ASPOD TAKES ADVANTAGE OF THE NODAL REGRESSION DUE TO THE
C OBLATENESS OF THE EARTH TO CHANGE LONGITUDINAL LOCATIONS
C *****
C
C IMPLICIT DOUBLE PRECISION (A-H,O-Z)
C
C DIMENSION RP(10),RA(10),AI(10),ARG(10),ZLONG(10),ANOM(10),A(10)
C DIMENSION ECC(10),P(10),PER(10),DELANOM(10),R1(2),R2(2),V1(2)
C DIMENSION V2(2),XANOM(10),C(100),YY(100),DV(50),SUM(50)
C REAL MASS(10)
C CHARACTER*64 FILENAME
C *****
C INPUT ROUTINE: INPUT PROGRAM CONTROL DATA AND ORBITAL PARAMETERS
C *****
C
C WRITE(*,*) ' '
C WRITE(*,*) ' A S P O D '
C WRITE(*,*) ' '
C WRITE(*,*) ' AUTONOMOUS SPACE PROCESSOR FOR ORBITAL DEBR
C *IS '
C WRITE(*,*) ' -----
C *-- '
C WRITE(*,*) ' '
C WRITE(*,*) ' '
C WRITE(*, '(A\)) ' - DO YOU WISH TO SAVE THE OUTPUT (Y/N)? '
C READ(*, '(A)') ANSWER
C IF (ANSWER.EQ.'Y'.OR.ANSWER.EQ.'y') THEN
C WRITE(*, '(A\)) ' - ENTER THE OUTPUT FILE NAME: '
C READ(*, '(A)') FILENAME
C OPEN (6,FILE=FILENAME,ACCESS='SEQUENTIAL',STATUS='NEW')
C ICHECK = 1
C END IF
C WRITE(*,*) ' '
C WRITE(*,*) ' '
C WRITE(*, '(A\)) ' 1 - WHAT IS THE DRY WEIGHT (Kg)? '
C READ(*,*) DRYWT
C WRITE(*, '(A\)) ' 2 - WHAT IS THE SPECIFIC IMPULSE OF THE TH
C *RUSTER? '
C READ(*,*) SPCFIMP
C WRITE(*, '(A\)) ' 3 - HOW MANY PIECES OF DEBRIS? '
C READ(*,*) N
C WRITE(*, '(A\)) ' 4 - WHAT IS THE INITIAL PARKING ORBIT ALTI
C *TUDE (KM)? '
C READ(*,*) PARKALT
C WRITE(*, '(A\)) ' 5 - WHAT IS THE FINAL PARKING ORBIT ALTITU
C *DE (KM)? '
C READ(*,*) FINALT
C WRITE(*, '(A\)) ' 6 - WHAT IS THE LOITER ORBIT ALTITUDE (KM)
C *? '

```

```

      READ(*,*) RLOIT
      WRITE(*, '(A\)' ) '      7 - WHAT IS THE APPROACH ORBIT DISTANCE (K
      *M)? '
      READ(*,*) APRCH
      WRITE(*, '(A\)' ) '      8 - WHAT IS THE FUEL TANK MASS PERCENTAGE?
      * '
      READ(*,*) PERC
C
      WRITE(*,*) ' '
99    WRITE(*, '(A\)' ) '      WOULD YOU LIKE TO MAKE A CHANGE (Y/N)? '
      READ(*, '(A)' ) YESNO
      IF (YESNO.EQ.'Y'.OR.YESNO.EQ.'y') THEN
        WRITE(*, '(A\)' ) '      WHAT NUMBER IS TO BE CHANGED? '
        READ(*,*) NUM
        IF (NUM.EQ.1) GO TO 501
        IF (NUM.EQ.2) GO TO 502
        IF (NUM.EQ.3) GO TO 503
        IF (NUM.EQ.4) GO TO 504
        IF (NUM.EQ.5) GO TO 505
        IF (NUM.EQ.6) GO TO 506
        IF (NUM.EQ.7) GO TO 507
        IF (NUM.EQ.8) GO TO 508
501    WRITE(*, '(A\)' ) '      1 - WHAT IS THE DRY WEIGHT (Kg)? '
        READ(*,*) DRYWT
        GO TO 99
502    WRITE(*, '(A\)' ) '      2 - WHAT IS THE SPECIFIC IMPULSE OF THE T
      *HRUSTER? '
        READ(*,*) SPCFIMP
        GO TO 99
503    WRITE(*, '(A\)' ) '      3 - HOW MANY PIECES OF DEBRIS? '
        READ(*,*) N
        GO TO 99
504    WRITE(*, '(A\)' ) '      4 - WHAT IS THE INITIAL PARKING ORBIT ALT
      *ITUDE (KM)? '
        READ(*,*) PARKALT
        GO TO 99
505    WRITE(*, '(A\)' ) '      5 - WHAT IS THE FINAL PARKING ORBIT ALTIT
      *UDE (KM)? '
        READ(*,*) FINALT
        GO TO 99
506    WRITE(*, '(A\)' ) '      6 - WHAT IS THE LOITER ORBIT ALTITUDE (KM
      *)? '
        READ(*,*) RLOIT
        GO TO 99
507    WRITE(*, '(A\)' ) '      7 - WHAT IS THE APPROACH ORBIT DISTANCE (
      *KM)? '
        READ(*,*) APRCH
        GO TO 99
508    WRITE(*, '(A\)' ) '      8 - WHAT IS THE FUEL TANK MASS PERCENTAGE
      *? '
        READ(*,*) PERC
        GO TO 99
      END IF
      WRITE(*,*) ' '
      WRITE(*,*) ' '
      WRITE(*,*) ' *** INPUT THE ORBITAL PARAMETERS IN THE ORDER OF REND
      *EZVOUS *** '
C
      DO 7 L = 1,N
        WRITE(*,*) ' '

```

```

1000 WRITE(*,1000)L
      FORMAT(5X,'TARGET #',I2/)
      WRITE(*,'(A\)' ) ' 1 - WHAT IS THE PERIGEE RADIUS (KM)? '
      READ(*,*) RP(L)
      WRITE(*,'(A\)' ) ' 2 - WHAT IS THE APOGEE RADIUS (KM)? '
      READ(*,*) RA(L)
      WRITE(*,'(A\)' ) ' 3 - WHAT IS THE ARGUMENT OF PERIGEE (DEG)
*? '
      READ(*,*) ARG(L)
      WRITE(*,'(A\)' ) ' 4 - WHAT IS THE INCLINATION (DEG)? '
      READ(*,*) AI(L)
      WRITE(*,'(A\)' ) ' 5 - WHAT IS THE RAAN (DEG)? '
      READ(*,*) ZLONG(L)
      WRITE(*,'(A\)' ) ' 6 - WHAT IS THE MASS (Kg)? '
      READ(*,*) MASS(L)
      WRITE(*,*) ' '
88    WRITE(*,'(A\)' ) ' WOULD YOU LIKE TO MAKE A CHANGE (Y/N)? '
      READ(*,'(A)' ) ANS
      IF (ANS.EQ.'Y'.OR.ANS.EQ.'y') THEN
        WRITE(*,'(A\)' ) ' WHAT NUMBER IS TO BE CHANGED? '
        READ(*,*)NNN
        IF (NNN.EQ.1) GO TO 1
        IF (NNN.EQ.2) GO TO 2
        IF (NNN.EQ.3) GO TO 3
        IF (NNN.EQ.4) GO TO 4
        IF (NNN.EQ.5) GO TO 5
        IF (NNN.EQ.6) GO TO 6
1      WRITE(*,'(A\)' ) ' 1 - WHAT IS THE PERIGEE RADIUS (KM)? '
        READ(*,*) RP(L)
        GO TO 88
2      WRITE(*,'(A\)' ) ' 2 - WHAT IS THE APOGEE RADIUS (KM)? '
        READ(*,*) RA(L)
        GO TO 88
3      WRITE(*,'(A\)' ) ' 3 - WHAT IS THE ARGUMENT OF PERIGEE (DEG
*)? '
        READ(*,*) ARG(L)
        GO TO 88
4      WRITE(*,'(A\)' ) ' 4 - WHAT IS THE INCLINATION (DEG)? '
        READ(*,*) AI(L)
        GO TO 88
5      WRITE(*,'(A\)' ) ' 5 - WHAT IS THE RAAN (DEG)? '
        READ(*,*) ZLONG(L)
        GO TO 88
6      WRITE(*,'(A\)' ) ' 6 - WHAT IS THE MASS (Kg)? '
        READ(*,*) MASS(L)
        GO TO 88
      END IF
7      CONTINUE
C
C *****
C
C      DEFINE CONSTANTS, WRITE DATA TO LU6, CHANGE ALL ANGLES TO RADIANS
C
C *****
C
C      PI = 4.0*ATAN(1.0)
C      ZMU = 398600.0
C      ALPHA = -1000.0/(SPCFIMP*9.81)
C      PERC = PERC/100.
C

```



```

IF (ICHECK.EQ.1) THEN
  WRITE(6,899)
  WRITE(6,900)
END IF
899  WRITE(*,899)
  FORMAT(25X,24HASPOD MISSION PARAMETERS)
  WRITE(*,900)
900  FORMAT(/5X,6HTARGET,3X,7HPERIGEE,3X,6HAPOGEE,4X,5HINCL.,5X,4HRAAN,
*      5X,4HARG.,5X,4HMASS/5X,62H-----)
*-----/)

C
DO 8 L = 1,N
  IF (ICHECK.EQ.1) THEN
    WRITE(6,901)L,RP(L),RA(L),AI(L),ZLONG(L),ARG(L),MASS(L)
  END IF
  WRITE(*,901)L,RP(L),RA(L),AI(L),ZLONG(L),ARG(L),MASS(L)
901  FORMAT(7X,I2,5X,F7.1,2X,F7.1,4X,F5.1,4X,F5.1,5X,F5.1,4X,F6.1/)
  A(L) = (RP(L)+RA(L))/2.0
  ECC(L) = 1.0-RP(L)/A(L)
  P(L) = A(L)*(1.0-ECC(L)**2)
  AI(L) = AI(L)*PI/180.0
  ARG(L) = ARG(L)*PI/180.0
  ZLONG(L) = ZLONG(L)*PI/180.0
8  CONTINUE

C
IF (ICHECK.EQ.1) THEN
  WRITE(6,902) DRYWT,SPCFIMP,PARKALT,FINALT,RLOIT,APRCH
END IF
902  WRITE(*,902) DRYWT,SPCFIMP,PARKALT,FINALT,RLOIT,APRCH
  FORMAT(/5X,'ASPOD DRY WEIGHT' = ',F7.1,3H Kg,
*      /5X,'SPECIFIC IMPULSE' = ',F7.1,5H SECS,
*      /5X,'PARKING ORBIT ALTITUDE' = ',F7.1,3H KM,
*      /5X,'FINAL ORBIT ALTITUDE' = ',F7.1,3H KM,
*      /5X,'LOITER ORBIT ALTITUDE' = ',F7.1,3H KM,
*      /5X,'APPROACH ORBIT DISTANCE' = ',F7.1,3H KM)

C
IF (ICHECK.EQ.1) THEN
  WRITE(6,903)
  WRITE(6,904)
END IF
903  WRITE(*,903)
  FORMAT(/27X,21HASPOD MISSION PROFILE/)
  WRITE(*,904)
904  FORMAT(15X,3HDV1,7X,3HDV2,7X,3HDV3,7X,3HDV4,7X,3HDV5,7X,3HDV6,/15X
*,54H-----)

C
C *****
C
C      CALCULATIONS FOR TRANSFER FROM PARKING ORBIT
C
C *****
C
C
C  PERI = 6372+PARKALT
C  APO = RA(1)
C  SEMIAX = PERI+APO
C  VTRAN1 = DSQRT(2.0*ZMU*(1.0/PERI-1.0/SEMIAX))
C  VPER = DSQRT(ZMU/PERI)
C  DV(1) = ABS(VTRAN1-VPER)
C  VTRAN2 = DSQRT(2.0*ZMU*(1.0/APO-1.0/SEMIAX))
C  VAPO = DSQRT(2.0*ZMU*(1.0/APO-1.0/(2.0*A(1))))

```

```

      DV(2) = ABS(VTRAN2-VAPO)
      IF (ICHECK.EQ.1) THEN
        WRITE(6,905) DV(1),DV(2)
      END IF
      WRITE(*,905) DV(1),DV(2)
905   FORMAT(/5X,4HPARK,3X,F7.4,3X,F7.4,7X,3H0.0,7X,3H0.0,7X,3H0.0,7X,3H
      *0.0)
C
C *****
C
C   CALCULATIONS FOR TRANSFERS BETWEEN PIECES OF DEBRIS
C
C *****
C
      RLOIT = RLOIT+6372.
      JJ = 3
      DO 10 I = 1,N-1
        AXIS = RA(I)+RLOIT
        VEL1 = SQRT(2*ZMU*(1./RA(I)-1./(2.*A(I))))
        VT1 = SQRT(2*ZMU*(1./RA(I)-1./AXIS))
        DV(JJ) = ABS(VEL1-VT1)
        VEL2 = SQRT(ZMU/RLOIT)
        VT2 = SQRT(2*ZMU*(1./RLOIT-1./AXIS))
        DV(JJ+1) = ABS(VEL2-VT2)
        AXIS = RA(I+1)+APRCH+RLOIT
        PERI = RA(I+1)+APRCH
        VT3 = SQRT(2*ZMU*(1./RLOIT-1./AXIS))
        DV(JJ+2) = ABS(VEL2-VT3)
        VEL4 = SQRT(ZMU/PERI)
        VT4 = SQRT(2*ZMU*(1./PERI-1./AXIS))
        DV(JJ+3) = ABS(VEL4-VT4)
        AXIS = RA(I+1)+PERI
        VT5 = SQRT(2*ZMU*(1./PERI-1./AXIS))
        DV(JJ+4) = ABS(VEL4-VT5)
        VEL6 = SQRT(2*ZMU*(1./RA(I+1)-1./(2.*A(I+1))))
        VT6 = SQRT(2*ZMU*(1./RA(I+1)-1./AXIS))
        DV(JJ+5) = ABS(VEL6-VT6)
        IF (ICHECK.EQ.1) THEN
          WRITE(6,1100) I,DV(JJ),DV(JJ+1),DV(JJ+2),DV(JJ+3),DV(JJ+4),DV(JJ+5
*)
          END IF
          WRITE(*,1100) I,DV(JJ),DV(JJ+1),DV(JJ+2),DV(JJ+3),DV(JJ+4),DV(JJ+5
*)
1100   FORMAT(/6X,1H#,I2,3X,F7.4,3X,F7.4,3X,F7.4,3X,F7.4,3X,F7.4,3X,F7.4
*)
          JJ = JJ+6
10     CONTINUE
C
C *****
C
C   CALCULATIONS FOR TRANSFER TO FINAL PARKING ORBIT
C
C *****
C
      M = 6*(N-1)+3
      PERIG = FINALT+6372.
      AXIS = RA(N)+PERIG
      VA1 = DSQRT(2.*ZMU*(1.0/RA(N)-1.0/AXIS))
      VA0 = DSQRT(2.*ZMU*(1.0/RA(N)-1.0/(2.*A(N))))
      DV(M) = ABS(VA1-VA0)

```

```

VP1 = DSQRT(2.*ZMU*(1.0/PERIG-1.0/AXIS))
VP0 = DSQRT(ZMU/PERIG)
DV(M+1) = ABS(VP1-VP0)
C
IF (ICHECK.EQ.1) THEN
  WRITE(6,907)N,DV(M),DV(M+1)
END IF
WRITE(*,907)N,DV(M),DV(M+1)
907  FORMAT(/6X,1H#,I2,3X,F7.4,3X,F7.4,7X,3H0.0,7X,3H0.0,7X,3H0.0,7X,3H
*0.0)
C
C *****
C
C   CALCULATE THE REQUIRED PROPELLANT MASS, FUEL TANK MASS, AND
C   TOTAL LAUNCH MASS.
C *****
C
SUM(1) = 0.0
LL = 6*(N-1)+4
DO 21 I = 1,LL
  SUM(1) = SUM(1)+DV(I)
21  CONTINUE
C(1) = 1.-EXP(ALPHA*SUM(1))
J = 3
DO 23 I = 2,N+1
  DO 22 K = J,LL
    SUM(I) = SUM(I)+DV(K)
22  CONTINUE
  J = J+6
  C(I) = 1.-EXP(ALPHA*SUM(I))
23  CONTINUE
PAYLD = 0.0
DO 30 JJ = 1,N
  PAYLD = PAYLD+C(JJ+1)*MASS(JJ)
30  CONTINUE
C
PROPEL = (C(1)*DRYWT+PAYLD)/(1.0-(1.0+PERC)*C(1))
WEIGHT = (1.+PERC)*PROPEL+DRYWT
TANK = PERC*PROPEL
C
IF (ICHECK.EQ.1) THEN
  WRITE(6,908)PROPEL,WEIGHT,TANK
END IF
WRITE(*,908)PROPEL,WEIGHT,TANK
908  FORMAT(/5X,'PROPELLANT MASS = ',F9.2,3H Kg,
*        /5X,'LAUNCH MASS      = ',F9.2,3H Kg,
*        /5X,'TANK MASS        = ',F9.2,3H Kg)
C
WRITE(*,*) ' '
STOP '      ASPOD ANALYSIS COMPLETED '
END

```

SAMPLE CALCULATIONS FOR PLANAR TRANSFERS
THROUGH USE OF NODAL REGRESSION - 2 TARGET
MISSION WITH LOITER ORBIT ALTITUDE OF 950 KM

TRANSFER FROM 500 KM PARKING ORBIT TO FIRST
RENDEREVUS @ 150 KM.

$$V_1 = \sqrt{\frac{\mu}{r_1}} \quad \mu = 3.986 \times 10^5 \text{ km}^3/\text{s}^2$$

$$r_1 = 500 + 6372 = 6872 \text{ km}$$

$$\therefore V_1 = 7.6160 \text{ km/s}$$

$$V_2 = \sqrt{\frac{\mu}{r_2}} \quad r_2 = 750 + 6372 = 7122 \text{ km}$$

$$\therefore V_2 = 7.4311 \text{ km/s}$$

TRANSFER ORBIT: $2a = 6872 + 7122 = 13994 \text{ km}$

$$V_{t1} = \sqrt{2\mu \left[\frac{1}{r_1} - \frac{1}{2a} \right]}$$

$$\therefore V_{t1} = 7.6337$$

$$V_{t2} = \sqrt{2\mu \left[\frac{1}{r_2} - \frac{1}{2a} \right]}$$

$$\therefore V_{t2} = 7.4140$$

THUS,

$$\Delta V_1 = V_{t1} - V_1 = 0.0677 \text{ km/s}$$

$$\Delta V_2 = V_2 - V_{t2} = 0.0671 \text{ km/s}$$

TRANSFER FROM FIRST DEBRIS ORBIT TO LOITER
ORBIT (950 KM).

$$V_3 = V_2 = 7.4811 \text{ km/s}$$

$$V_4 = \sqrt{\frac{\mu}{r_4}} \quad r_4 = 950 + 6372 = 7322 \text{ km}$$

$$\therefore V_4 = 7.3733 \text{ km/s}$$

$$\text{TRANSFER KEET: } 2a = 7322 + 7122 = 14444 \text{ km}$$

$$V_{t3} = \sqrt{2\mu \left[\frac{1}{r_3} - \frac{1}{2a} \right]} \quad r_3 = r_2 = 7122 \text{ km}$$

$$\therefore V_{t3} = 7.5328 \text{ km/s}$$

$$V_{t4} = \sqrt{2\mu \left[\frac{1}{r_4} - \frac{1}{2a} \right]}$$

$$\therefore V_{t4} = 7.3270 \text{ km/s}$$

THUS,

$$\Delta V_3 = V_{t3} - V_3 = 0.0517 \text{ km/s}$$

$$\Delta V_4 = V_4 - V_{t4} = 0.0513 \text{ km/s}$$

TRANSFER FROM OUTER ORBIT TO SECOND DEBRIS ORBIT (750 km).

$$\Delta V_5 = \Delta V_4 = 0.0513 \text{ km/s}$$

$$\Delta V_6 = \Delta V_3 = 0.0517 \text{ km/s}$$

TRANSFER TO FINAL PARKING ORBIT (500 km)

$$\Delta V_7 = \Delta V_2 = 0.0671 \text{ km/s}$$

$$\Delta V_8 = \Delta V_1 = 0.0677 \text{ km/s}$$

$$\text{MASS CALCULATIONS: } I_{sp} = 325 \text{ sec} \quad g = 9.81 \text{ m/s}^2$$

$$M_{dry} = 526 \text{ kg}$$

$$\text{TOTAL } \Delta V (\Delta V_T) = 0.4756 \text{ km/s}$$

TOTAL ΔV AFTER FIRST RENDEZVOUS (ΔV_{T1})

$$\Delta V_{T1} = 0.3403 \text{ km/s}$$

TOTAL ΔV AFTER SECOND RENDEZVOUS (ΔV_{T2})

$$\Delta V_{T2} = 0.1343 \text{ km/s}$$

$$C_1 = 1 - \exp \left[\frac{-\Delta V_{T1}}{T_{sp} \cdot g} \right] = 0.1336$$

$$C_2 = 1 - \exp \left[\frac{-\Delta V_{T1}}{T_{sp} \cdot g} \right] = 0.1014$$

$$C_3 = 1 - \exp \left[\frac{-\Delta V_{T2}}{T_{sp} \cdot g} \right] = 0.0414$$

$$\therefore M_p = \frac{C_1 M_{Dry} + (C_2 + C_3) M_{SBSIS}}{1 - 1.023 C_1}$$

$$\underline{M_p = 417.7 \text{ Kg}}$$

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DERIVATION OF NODAL REGRESSION PER DAY
FOR CIRCULAR ORBITS.

$$\frac{d\Omega}{dt} = - \frac{3\mu_0 J_2 R_0^2}{r^4 V_0} \sin \theta \cos i \quad \text{KAPLAN, P 356}$$

WHERE :

$$\mu_0 = 3.986 \times 10^5 \text{ km}^3/\text{s}^2$$

$$R_0 = 6378 \text{ km}$$

$$J_2 = 1082.7 \times 10^{-6} \quad \text{KAPLAN, P 282}$$

FOR CIRCULAR ORBITS, $r(t) = \text{CONST}$
 $V_0(t) = \text{CONST}$

$$V_0 = \sqrt{\frac{\mu_0}{r}}$$

$$\theta(t) = \frac{V_0}{r} t = \frac{\sqrt{\mu_0}}{r^{3/2}} t$$

$$\therefore \frac{d\Omega}{dt} = - \frac{3\mu_0^{1/2} R_0^2 J_2}{r^{7/2}} \sin \left\{ \frac{\mu_0^{1/2}}{r^{3/2}} t \right\} \cos i$$

$$\Delta\Omega = \int \frac{d\Omega}{dt} dt = - \frac{3\mu_0^{1/2} R_0^2 J_2}{r^{7/2}} \cdot \frac{r^{3/2}}{\mu_0^{1/2}} \cos i \left\{ \frac{\mu_0^{1/2}}{2r^{3/2}} t - \frac{1}{4} \sin \left[\frac{2\mu_0^{1/2}}{r^{3/2}} t \right] \right\}$$

$$\text{FOR } \Delta\Omega / \text{DAY} \quad t = 3600 \frac{\text{SEC}}{\text{HR}} \times 24 \text{ HRS} = 86400 \text{ SEC}$$

$$\therefore \Delta\Omega = \frac{132129}{r^2} \cos i \left[\frac{27274233}{r^{3/2}} - \frac{1}{4} \sin \left\{ \frac{1090968947}{r^{3/2}} \right\} \right]$$

(RAD/DAY)

$$\text{EXAMPLE : } r = 750 + 6378 = 7128 \text{ km}$$

$$i = 35^\circ$$

$$\Delta\Omega = 0.097 \text{ RAD/DAY} = 5.56 \text{ DEG/DAY}$$

$$\text{@ } r = 950 + 6378 = 7328 \text{ km}$$

$$\Delta\Omega = 0.038 \text{ RAD/DAY} = 2.17 \text{ DEG/DAY}$$

APPENDIX B

POWER BUDGET AND SOLAR ARRAY SIZING ANALYSIS

aspod power budget

	A	B	C	D	E	F	G
1	ITEM		PEAK PWR, W		DUTY CYCLE %		AVERAGE, W
2							
3	power interface unit		1		10		0.1
4	sc command		2		100		2
5	data storage		2.5		100		2.5
6	sensors/interfaces		2		100		2
7	sc housekeeping		2		100		2
8	communications		20		10		2
9	robotics		100		10		10
10	radar		30		50		15
11							
12	subtotal:		159.5			subtotal:	35.6
13	40% contingency/uncertainty		63.8			40% conting.	14.24
14	total		223.3			total	49.84

Solar Array Sizing

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- Support 50 W average power
- Worst case of 27% Sunlight during orbit

(time in sun) $t_{\text{sun}} = 1554$ seconds

(time occulted) $t_{\text{dark}} = 4240$ seconds

(orbital period) Period = 5794 seconds (97 min.)

Collector area:

$$A_c = \frac{L}{\phi_s \eta_p \eta_c \eta_d} \left(\frac{BOL}{EOL} \right) \left[1 + \frac{t_{\text{dark}}}{\eta_s t_{\text{sun}}} \right] \quad (1)$$

terms:

$L = 50$ W (load)

$\phi_s = 1353$ W (solar constant)

$\eta_p = .18$ (photovoltaic efficiency)
% sun captured by cells

$\eta_c = .7$ (collection efficiency - array geometry,
cell spacing, array orientation)

$\left(\frac{BOL}{EOL} \right) = 1.2$ (degradation from beginning of life-BOL
to end-EOL)

$\eta_d = .9$ (electrical processing efficiency)

$\eta_s = .6$ (battery storage efficiency $\frac{\text{energy out}}{\text{energy in}}$)

$$\text{thus, } A_c = \frac{50}{1353 (.18)(.7)(.7)} (1.7) \left[1 + \frac{4240}{.6 (1554)} \right]$$

$$\underline{A_c = 1.68 \text{ m}^2} \quad (\text{per } 50 \text{ w load})$$

- this corresponds to an array power of:

$$A_c * \phi_s = 1.68 (1353) = 2282 \text{ watts}$$

- for a specific power of $190 \text{ w/kg}^{(2)}$,

$$2282/190 = \underline{12 \text{ kg}} \text{ array mass.}$$

References:

- ① Woodcock, Gordon R., Space Stations and Platforms, Orbit Book Company, Malabar, FL, 1986, pp. 81-86.
- ② Hord, Michael, CRC Handbook of Space Technology: Status and Projections, CRC Press, Boca Raton, FL, 1985, pp. 53-56.

Computer program for percentage of time spent in sunlight per orbit.

Percentage of sunlight is dependant on the following parameters:

- Inclination of satellite orbit.
- Declination of Earth axis to the ecliptic plane.
- Longitude of ascending node of satellite orbit.

Resulting plot is from output data from program for declinations of 23.5° , 0° , and -23.5° .

$$(-1, 0) \cup (0, 1) = (-1, 1) \cup \{0\}$$

REPORT OF
OF DATA QUALITY

```

32 1  FORMAT(' THE MINIMUM=',F4.1,'%',/, ' THE MAXIMUM=',F4.1,'%',/,
    1  ' THE AVERAGE=',F4.1,'%',/)
33 1  WRITE(21,33)MIN,MAX,AVERG
    1  FORMAT(' THE MINIMUM=',F4.1,'%',/, ' THE MAXIMUM=',F4.1,'%',/,
    1  ' THE AVERAGE=',F4.1,'%',/)

WRITE(6,*) 'THE MINIMUM SUNSHINE ORBIT:'
WRITE(6,41)MTSUN,MTSHADOW,MZD,MOD
41 1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1,
    1  ' SEC',/, ' DELINATION=',F7.1,' DEG',/, ' OMEGA=',F7.1,' DEG',/)
WRITE(21,*) 'THE MINIMUM SUNSHINE ORBIT:'
42 1  WRITE(21,42)MTSUN,MTSHADOW,MZD,MOD
    1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1,
    1  ' SEC',/, ' DELINATION=',F7.1,' DEG',/, ' OMEGA=',F7.1,' DEG',/)

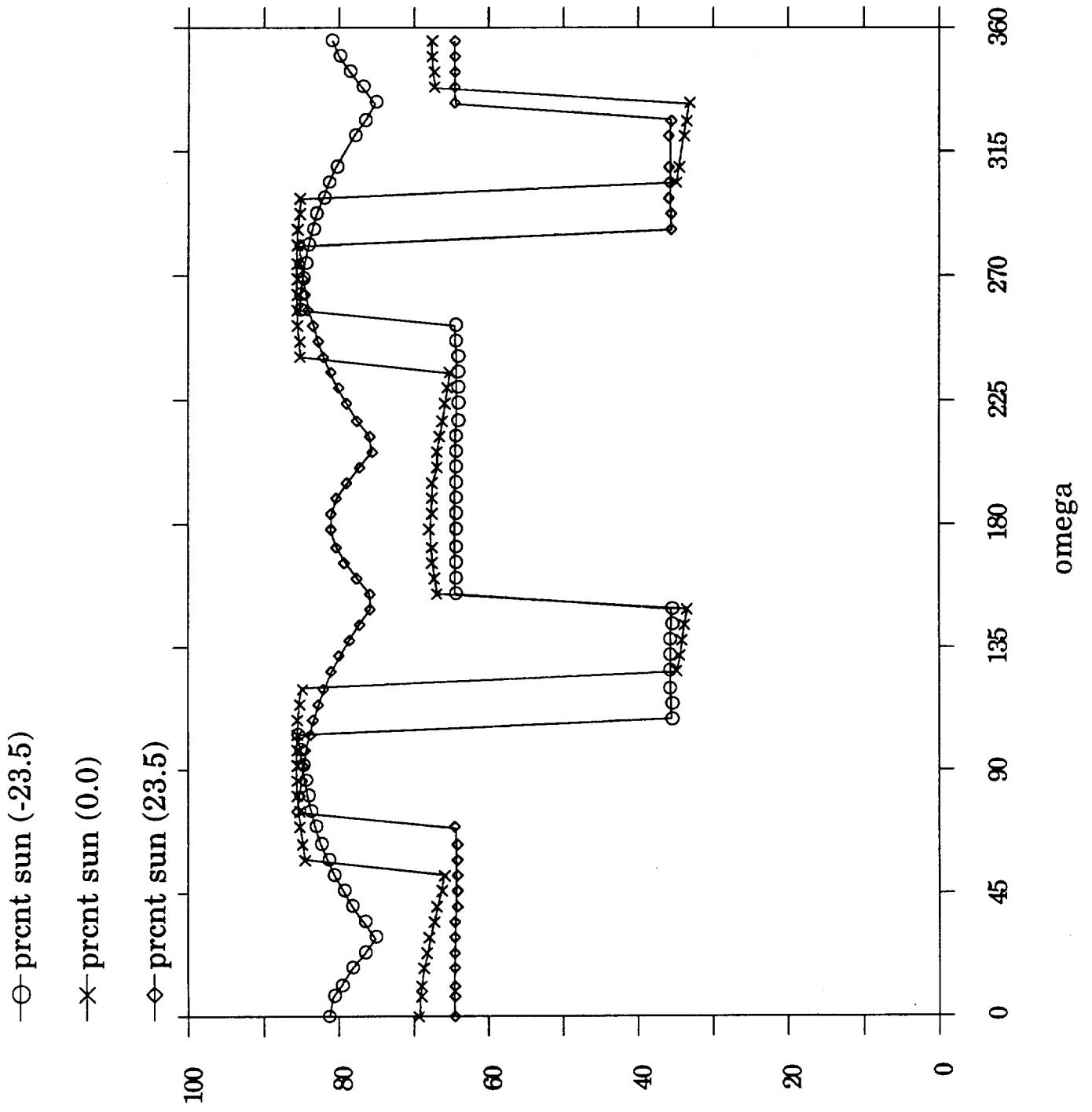
WRITE(6,*) 'THE MAXIMUM SUNSHINE ORBIT:'
WRITE(6,43)MXTSUN,MXTSHADOW,MXZD,MXOD
43 1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1,
    1  ' SEC',/,2X,' DELINATION=',F7.1,' DEG',/,2X,' OMEGA=',F7.1,' DEG')
WRITE(21,*) 'THE MAXIMUM SUNSHINE ORBIT:'
44 1  WRITE(21,44)MXTSUN,MXTSHADOW,MXZD,MXOD
    1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1,
    1  ' SEC',/,2X,' DELINATION=',F7.1,' DEG',/,2X,' OMEGA=',F7.1,' DEG')
    1  TSUN=AVERG*PERIOD/100
    1  TSHADOW=PERIOD-TSUN

WRITE(6,*) 'THE AVERAGE SUNSHINE ORBIT:'
WRITE(6,47)TSUN,TSHADOW
47 1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1)
    1  WRITE(21,*) 'THE AVERAGE SUNSHINE ORBIT:'
    1  WRITE(21,47)TSUN,TSHADOW
48 1  FORMAT(2X,' TIME SUN=',F7.1,' SEC',/,2X,' TIME SHADOW=',F7.1)

```

STOP
END

*23456789012345678921234567893123456789412345678951234567896123456789712



APPENDIX C

PROPULSION SYSTEM SELECTION DATA

Available Thruster Systems

1. Inert Gas - High pressure gas (7-31 MPa) reduced in pressure and expelled through nozzle. Typically includes propellant tank, fill valve, start valve, filter, regulator, low pressure relief valve, two pressure transducers, control valves and nozzles.
2. Tridyne Gas - Similar to Inert Gas system, but consists of 85% nitrogen and a stoichiometric mixture of hydrogen and oxygen. Thruster contains a catalyst causing oxygen and hydrogen to combine exothermically thus increasing gas temperature.
3. Hydrazine Direct Catalyst - Hydrazine passes through a catalyst bed and spontaneously initiates decomposition. Combustion products are expelled from catalyst bed through nozzle.
4. Hydrazine Resistojet - Similar to Hydrazine Direct Catalyst except decomposition is initiated by an electrically heated resistance element in thrust chamber.
5. Hydrazine Plenum - Similar, also, to Hydrazine Direct Catalyst; however, gases generated by decomposition are stored in a plenum for later gas expulsion.
6. Bipropellant MMH/N₂O₄ - Monomethyl hydrazine combines with nitrogen tetroxide spontaneously igniting. Combustion products are expelled through a nozzle.
7. Electrolysis - Low-molecular weight, gaseous propellant produced from electrolysis of water; gas is then ignited and expelled through nozzle.
8. Vaporizing Liquid - Liquid propellant is vaporized using a heater, and the vapor is expelled through a nozzle.
9. Subliming Solid - Solid propellant is heated, becomes pressurized by vapor pressure, and vapor is expelled through a nozzle.
10. Ion Thruster - Propellant (mercury or cesium) is ionized using anode-cathode arrangement; ions are accelerated in an electrostatic field and neutralized as they are emitted.
11. Pulsed Plasma - Propellant is ionized by a high voltage discharge and accelerated by the interaction of the discharge current with its own magnetic field.
12. Resistojet - Subsystem used with inert gases and vaporizing liquids; heats gases before expulsion using an electrical resistance heater.
13. Radioisojet - Similar to resistojet except gases are heated using a nuclear heater source.

Table A-1. Performance Characteristics of Propulsion Systems

Systems	Thrust (N)	Isp (lbf-s/lbm)
Inert Gas (80 deg. F)		
H ₂	0.000445-4.45	272
He	0.000445-4.45	165
Ne	0.000445-4.45	75
N ₂	0.000445-4.45	72
A	0.000445-4.45	52
Tridyne Gas (978 K)	0.445	143
Hydrazine Direct Catalyst		
Steady State Thrust	2.22-22.2	225
Cold Pulse	2.22-22.2	110
Hot (50th) Pulse	2.22-22.2	210
Hydrazine Resistojet		
Initial Pulse and Steady State	0.0445/0.445	175/200
Initial Pulse	0.445-22.2	210
Hydrazine Plenum		
Cold (300 K)	0.0445-0.267	100
Bipropellant		
MMH/N ₂ O ₄	2.22-22.2	289
Electrolysis		
Hot O ₂ /H ₂ Gas	0.222-22.2	350
Vaporizing Liquid		
Ammonia	4.448E-5 to 0.222	97
Freon	4.448E-5 to 0.222	52
Subliming Solid (300 K)		
Ammonium hydrosulfide	0.0445 0.00445	78 75
Ion Thruster	4.45E-6 to 0.00445	3000-7300
Pulsed Plasma	4.45E-6 to 44.5E-6	1000-4000
Resistojets (1366.5 K)		
Vaporizing Liquid (Ammonia)	0.0445	230
Inert Gas (H ₂)	4.45E-4 to 4.45	550
Inert Gas (N ₂)	4.45E-4 to 4.45	150
Radioisotjet (1366.5 K)		
Vaporizing Liquid (Ammonia)	0.0222-0.089	250



Ford Aerospace &
Communications Corporation

MODEL R-6C - 5-LBf (22N) BIPROPELLANT ROCKET ENGINE



Dimensional Characteristics

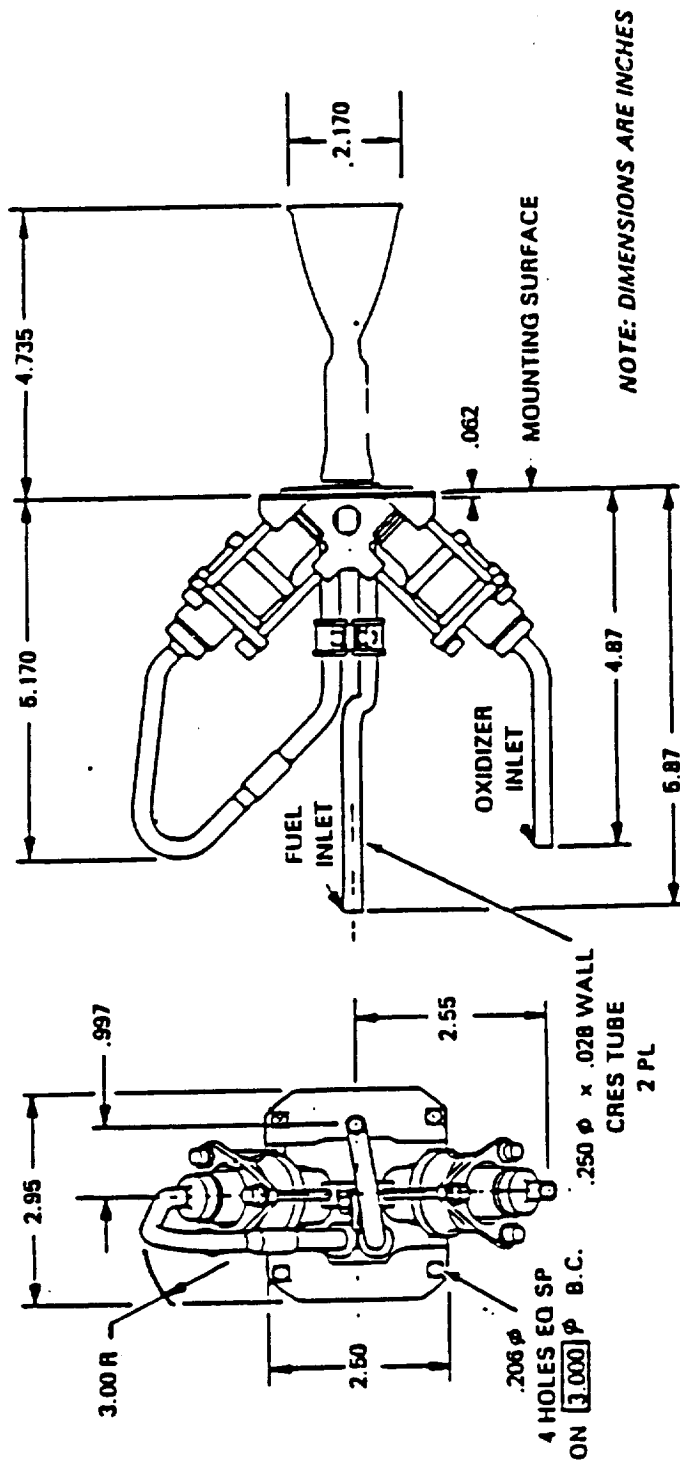
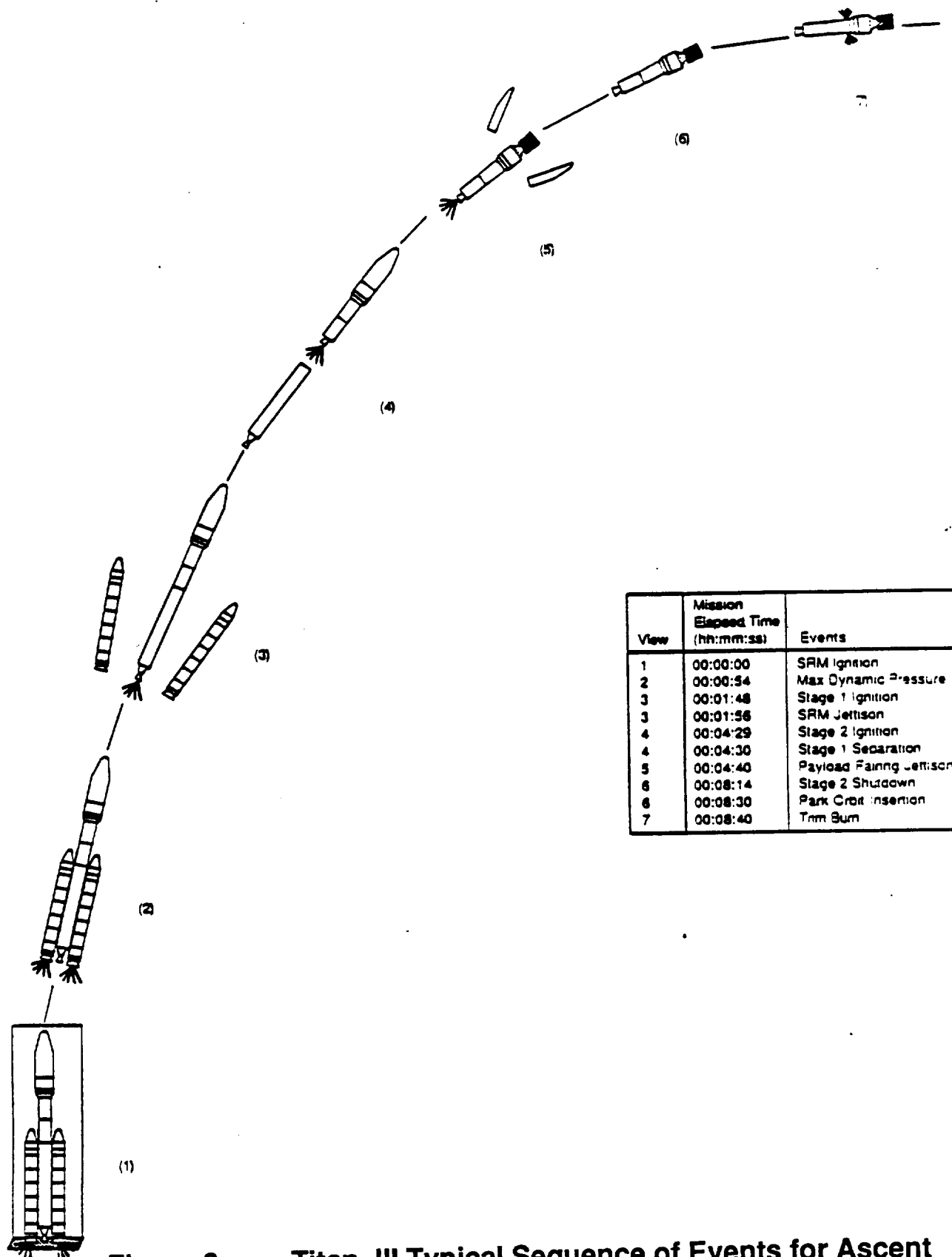


Figure 8

APPENDIX D
LAUNCH VEHICLE INTEGRATION DATA

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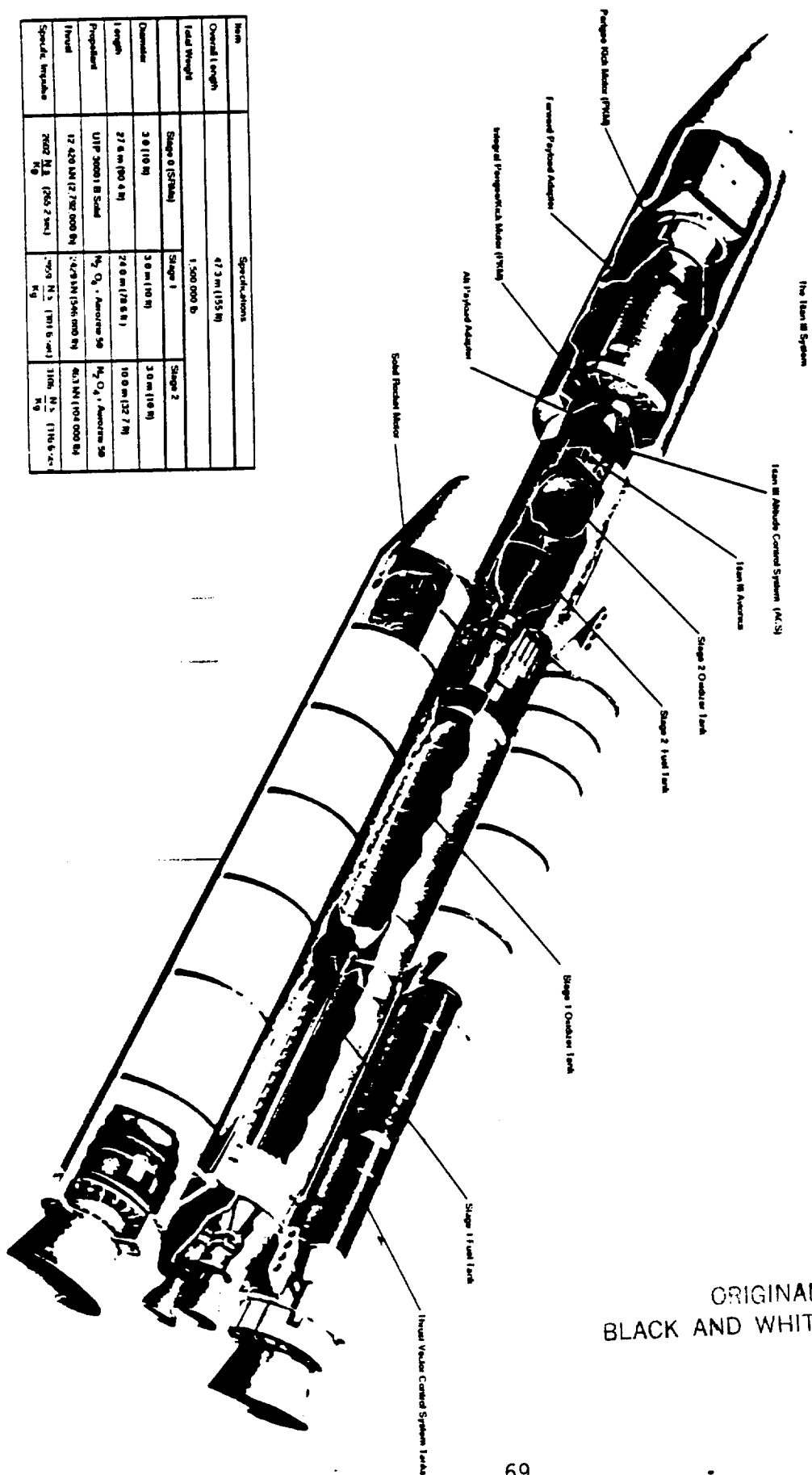
View	Mission Elapsed Time (hh:mm:ss)	Events
1	00:00:00	SRM Ignition
2	00:00:54	Max Dynamic Pressure
3	00:01:48	Stage 1 Ignition
3	00:01:56	SRM Jettison
4	00:04:29	Stage 2 Ignition
4	00:04:30	Stage 1 Separation
5	00:04:40	Payload Fairing Jettison
6	00:08:14	Stage 2 Shutdown
6	00:08:30	Park Orbit Insertion
7	00:08:40	Trim Burn

Figure 3

Titan III Typical Sequence of Events for Ascent Phase (LEO Mission)

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Item		Specifications	
Overall length		47.3 m (155 ft)	
Load Weight		1,500,000 lb	
	Stage 0 (S2700)	Stage 1	Stage 2
Diameter	3.0 (10 ft)	3.0 m (10 ft)	3.0 m (10 ft)
Length	27.0 m (90 ft)	24.0 m (79 ft)	10.0 m (33 ft)
Propellant	UHP 30001 B Solid	H_2 , O_2 - Ammonia 50	H_2 , O_2 - Ammonia 50
Thrust	12,420 kN (2,792,000 lb)	2,478 kN (556,000 lb)	463 kN (104,000 lb)
Specific Impulse	2602 N/A (265.2 sec)	2602 N/A (265.2 sec)	2602 N/A (265.2 sec)



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Figure 1 Titan III Low-Earth Orbit (LEO) Configuration

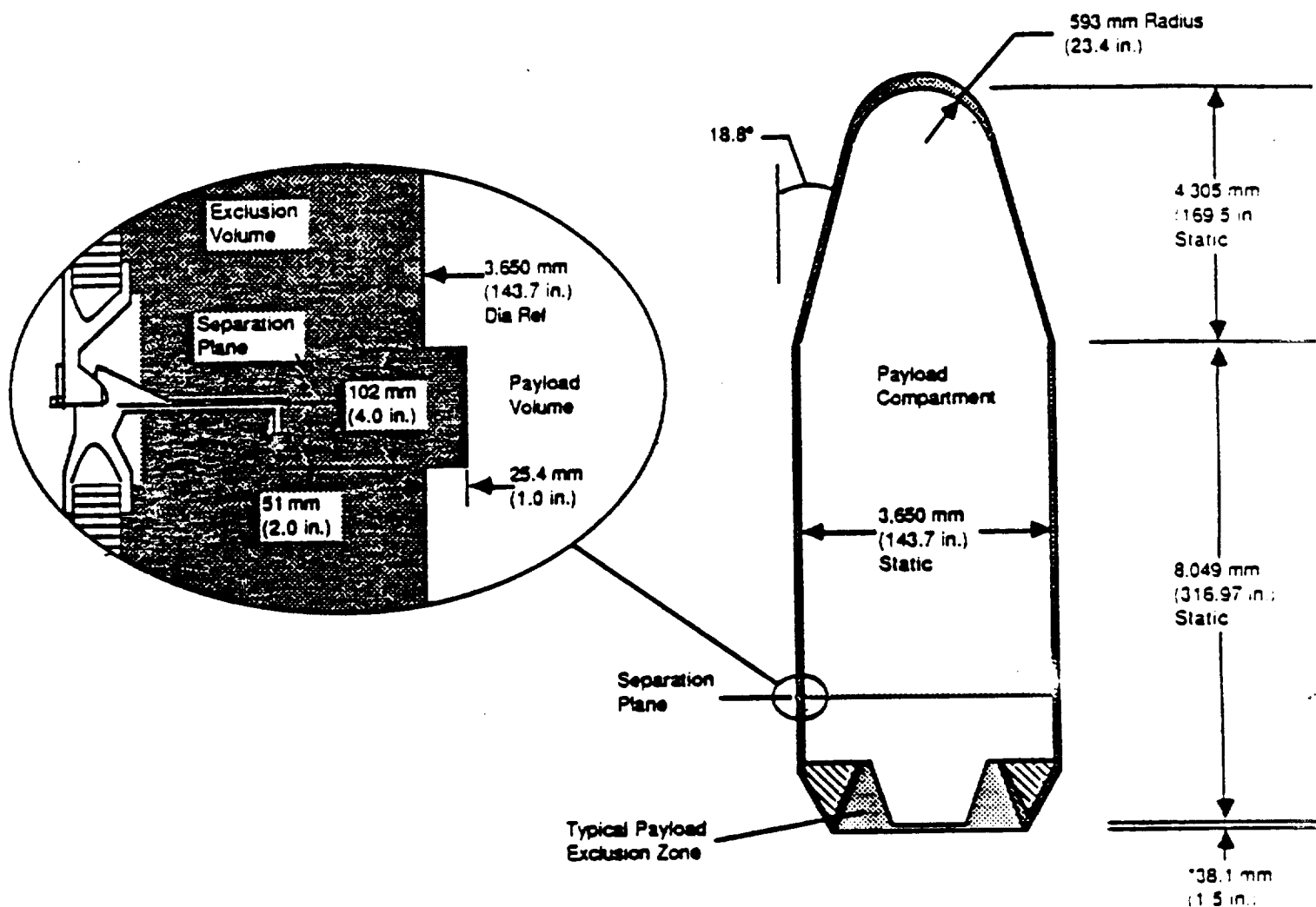


Figure 5 Single Payload Carrier Static Envelope

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10



Benilde-St. Margaret's

A Catholic Co-Educational High School

February 26, 1989

Kumar Ramoholli
Mechanical Engineer
Department of Aerospace
University of Arizona
Tucson, AZ 85721

Dear Kumar Ramoholli,

My name is Erin Cashin and I go to Benilde-St. Margaret's High School in Minneapolis, Minnesota. I'm in the tenth grade and I take biology. Our class has been doing projects where you find an article with biological basis and you write a summary on it. For one of my papers I wrote on an article taken from a December 1988 Discover magazine. The article was called "Sweeping up Space Junk". I was wondering what would happen if this robot didn't get all of the space junk? If it didn't would it be very dangerous? Could you please send me some more information on this?

Thank you,

Erin Cashin

Erin Cashin

Erin Cashin
Benilde-St. Margaret's
c/o Mr Mark Peterson
2501 Hwy 100 So.
St. Louis Park, MN. 55416

BON

815 W. Fifth St.
Loveland, Co. 80537

October 10, 1988

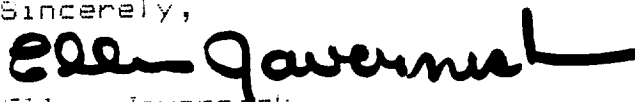
Aerospace Engineering Department
University of Arizona
Tucson, Arizona

Dear Mr. Ramohalli,

I was interested to read about the space vacuum cleaner that you and your students recently demonstrated to NASA. As a children's writer, I am always on the lookout for new discoveries that will appeal to young people.

Would it be possible for you to send more information about your research grant and your tidy robots. Editors often ask for photographs to accompany the stories they buy. If you have any such pictures available, I'd appreciate receiving those as well as the written information.

Sincerely,



Ellen Javernick

Chris Tulloch
8a London Road Twickenham
Middlesex TW1 3RR
01 892 5008

Kumar Ramohalli
Aerospace & Mechanical
Engineering Dept.,
Building I6,
University of Arizona,
Tucson 85721.

24 February 1989

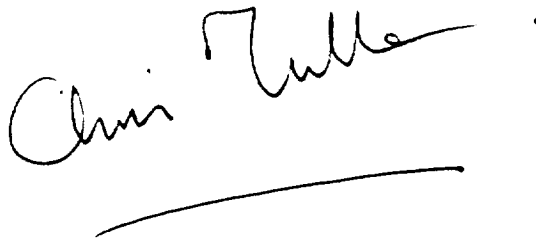
Dear Mr. Ramohalli,

You may be wondering why I rang from England,
but I was interested by your space robot which was
featured on a BBC-TV science programme "Tomorrows
World".

Are there any sketches or photographs of a
model you could let me have? This is in connection
with a project I am researching on pollution.

Look forward to hearing from you,

Yours sincerely,


A handwritten signature in cursive script, reading 'Chris Tulloch', followed by a horizontal line.

FREE RADICAL DONORS FOR FLAME AUGMENTATION
IN SUPPORT OF THE
NATIONAL AEROSPACE PLANE

Gordon Ingmire
David Andrus

For
NASA/USRA
Summer Conference
June 12,1989

INTRODUCTION

The purpose of this experiment is to determine the possible benefits of injecting free radical sources or donors into high speed flows in order to maximize the speed at which the flame 'blows out.' The initial information is a calibration of 'blow out' velocities without the injection of the donors. This was done in order to aid in the determination of the benefits of these injections.

PROCEDURE

The initial data were compiled by first determining the relationships between the gage pressures, volume flow rates, and velocities of both the fuel and oxidizer. The chosen fuel was methane, and a 60-40 Nitrogen-Oxygen mixture was used as the oxidizer. Two concentric pipes, with an effective L/D of approximately 150 were used in order to diminish any upstream disturbances.

Problems arose with non-concentricity of the pipes causing the flame to be unsymmetrical, and premature 'blow out' to occur. Once this problem was corrected, flame velocities and corresponding Reynold's Numbers were easily determined. The next step in the experiment is to obtain higher pressures which will give higher velocities without the use of the injection of free radicals.

Injection of the free radicals must be accomplished at a constant flow rate. This will be done using an IV bag filled with a 3% solution of Hydrogen Peroxide, which will produce free radicals. This injection should allow the flame to be held at higher velocities.

CALCULATIONS

Readings were taken of the Tank Static Pressure, P_t , Volumetric Scale Reading, SR , Laboratory Temperature T_{lab} , and Static Gage Pressure (at the flow meter), P_g for both the methane and oxidizer. The Scale Readings were then converted to Volume Flow Rates, V_a , using the Correlated Flow Table by Gilmont Instruments Inc. A corrected Volume Flow Rate, V_c , was then calculated using equation 1 which was derived using the perfect gas assumption.

$$V_c = V_a * [(P_g + 13.6) * 520 / (14.7 * T_{lab})]^{1/2} \quad \text{(eqn 1)}$$

The Velocity, U was determined by equation 2,

$$U = V_c / A \quad \text{(eqn 2)}$$

where A is the equivalent oxidizer flow area. The Reynold's Number for the oxidizer was calculated using the following relation:

$$Re = (P_g + 13.6) * U * D / (\mu R T) \quad \text{(eqn 3)}$$

where D is the effective diameter of 0.31225 inches, μ is the dynamic viscosity of $1.087 (10^{-5})$ lb/ft-s, R is the gas constant of 52.21 ft-lbf/lbm-R, and T is the absolute room temperature.

RESULTS

Actual 'blow out' velocities have not been determined at this time due to problems with the apparatus. Velocities and Reynold's Numbers at varying Tank Pressures have been calibrated and are shown in Table 1. Graphs have also been plotted to show the relationships of Velocity vs. Tank Pressure (Fig. 1), Gage Pressure vs. Tank Pressure (Fig. 2), Velocity vs. Volume Flow Rate (Fig. 3), Reynold's Number vs. Tank Pressure (Fig. 4), and Velocity vs. Reynold's Number (Fig. 5).

SUMMARY

As of this time, a Mach Number of approximately 0.5 is the maximum that can be reached. Due to this constriction, a regulator which can reach higher pressures is being sought in order to increase the velocity. Another recurring problem is the cracking of the volume flow tube. This appears to be due to the possibilities of reaching maximum allowable stress pressures of the tube. Hopefully this can be overcome by increasing the flow orifice diameter in the new regulator, allowing the pressure to decrease while achieving the desired flow rate.

Figure 3

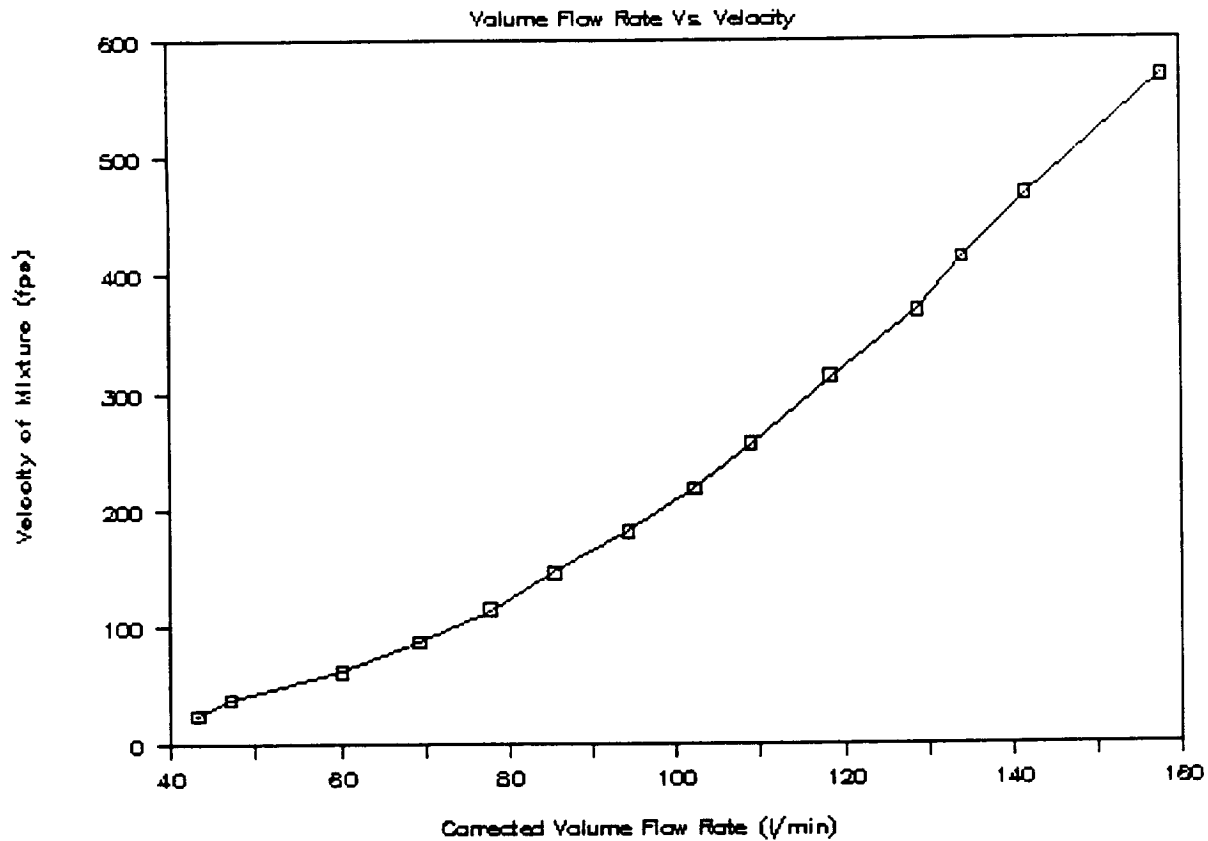


TABLE 1

Mix

Pt	Pg	Vr	Vc	u	Re
40	8.0	35.7	43.07	25.34	6754
50	11.0	36.5	47.00	38.01	11540
60	14.0	44.0	60.01	61.77	21041
70	17.0	48.2	69.22	86.52	32675
80	20.0	51.6	77.65	114.19	47350
90	23.2	54.2	85.35	145.60	66129
100	26.1	57.7	94.38	181.12	88743
110	29.0	60.3	102.17	217.86	114541
120	32.0	62.1	108.86	256.14	144152
130	36.0	64.8	118.47	313.60	191969
140	39.0	68.4	128.78	369.30	239735
150	42.0	69.3	134.14	414.27	284267
160	45.0	71.2	141.49	468.17	338588
175	49.0	76.8	157.74	568.34	439087

Methane

Pt	Pg	Vr	Vc	u
10	2.0	8.0	8.16	1.20
20	2.5	18.5	19.27	3.54
30	3.0	25.1	26.55	5.86
40	4.0	30.0	32.67	9.61
50	5.0	35.7	39.97	14.69
60	6.0	40.7	46.78	20.64

Figure 2

Pressure Relationship

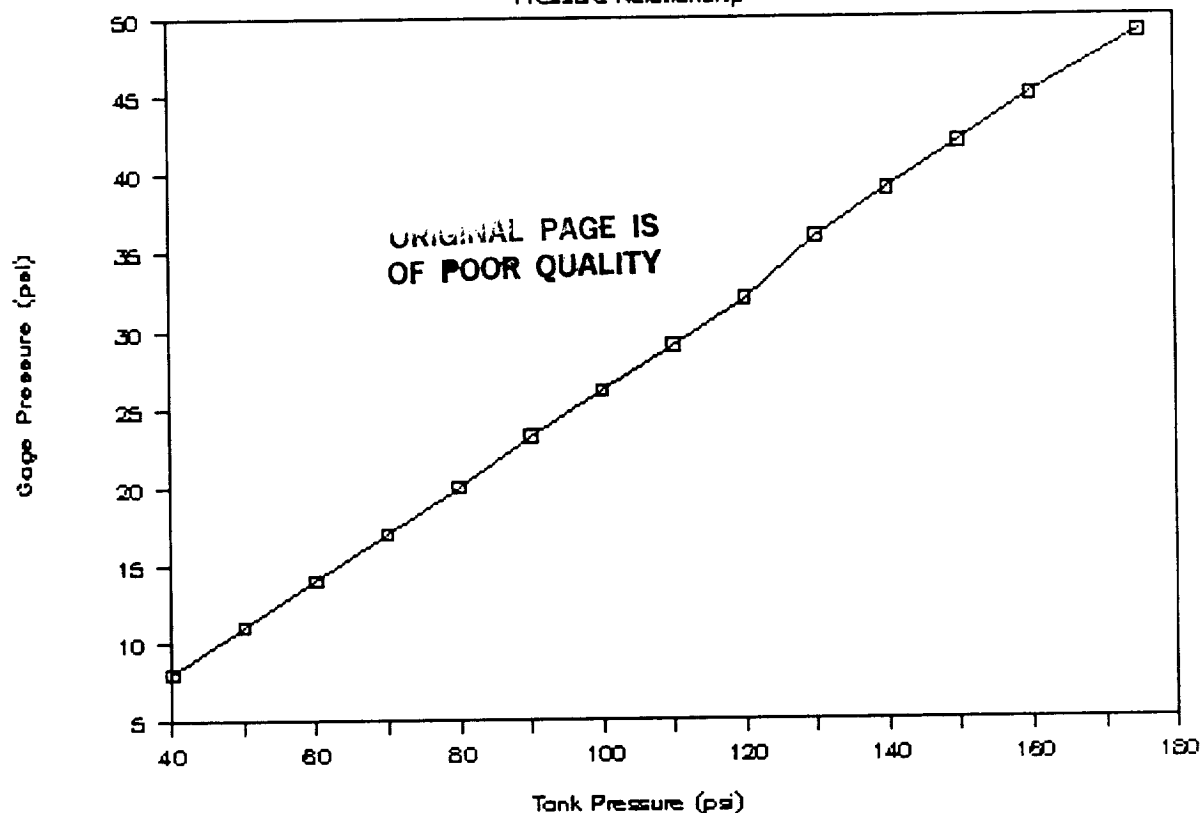


Figure 4

Tank Pressure Vs. Reynold's Number

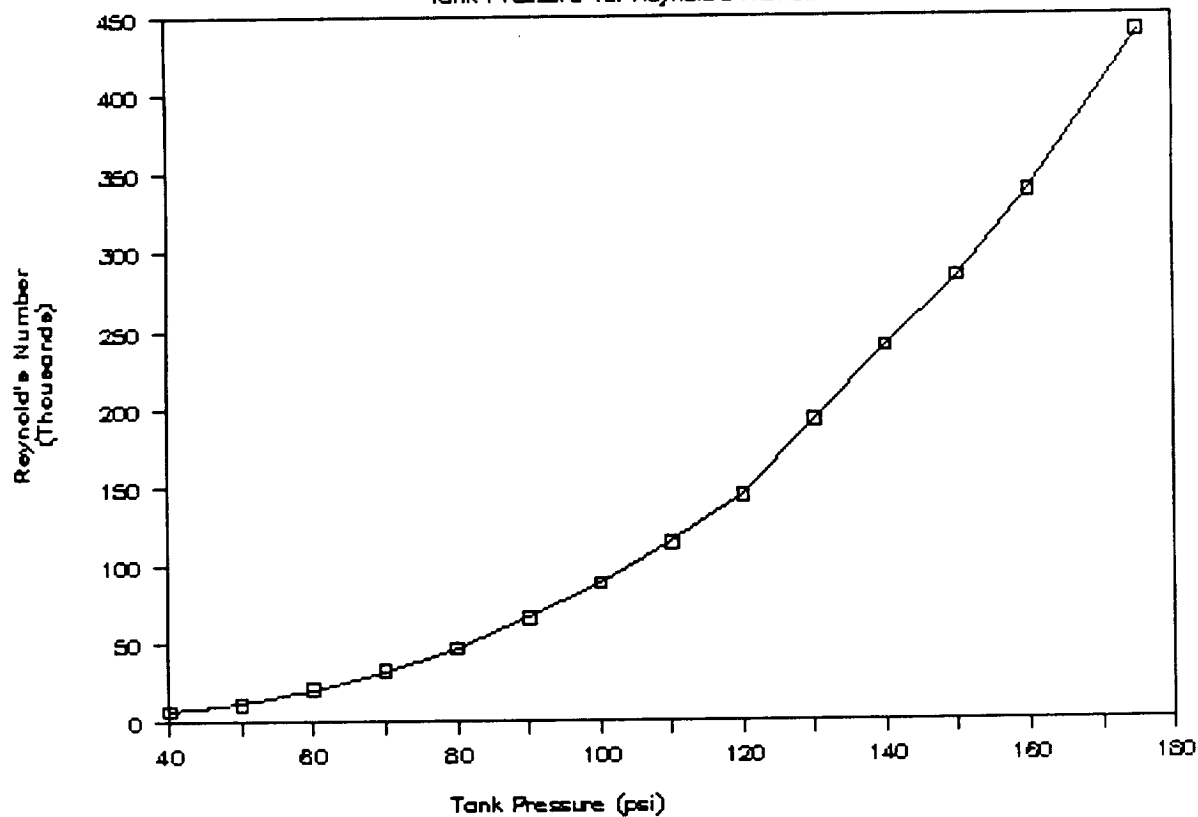


Figure 1

Velocity Vs. Pressure

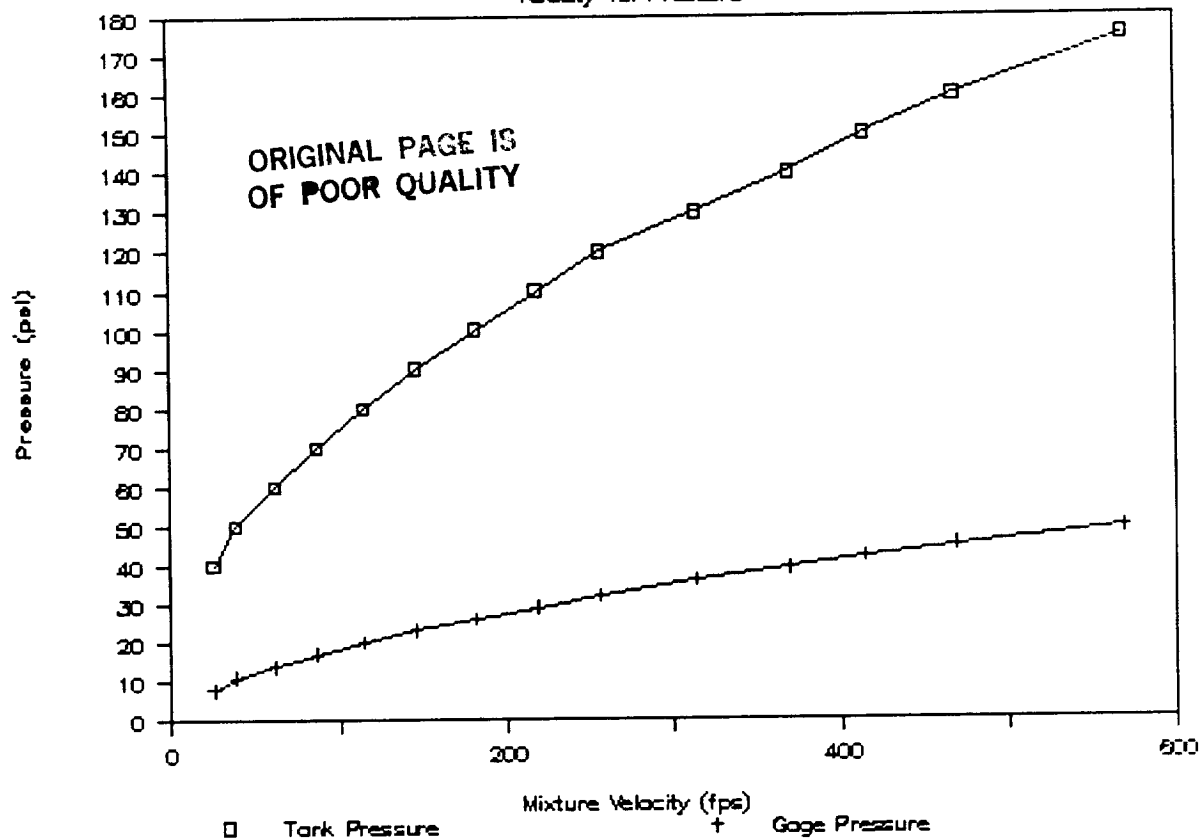


Figure 5

Velocity Vs. Reynold's Number

